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PROJECT APOLLO
SPACECRAFT DEVELOPMENT
STATEMENT OF WORK

PART 3
TECHNICAL APPROACH

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MANNED SPACECRAFT CENTER

Langley Air Force Base, Virginia

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3.1 INTRODUCTION.- The Technical Approach Section presents a technical description of the operational and flight plans and systems approach for the Apollo Spacecraft. The description constitutes a technical framework within which the initial design and operational modes of the Spacecraft are to be further developed.

The Apollo Spacecraft, operational, and flight plans described herein are defined by the requirements of the ultimate mission, lunar landing, and return. The resulting basic systems are then considered to be off-loaded for intermediate missions and qualification flights.

Several techniques for effecting earth and/or lunar-launch can be considered depending on the capabilities of various launch vehicles and operational know-how. The Spacecraft System described is designed for direct earth-launch, lunar-landing, and lunar-launch; but, is intended to be suitable for use with any of a variety of earth-launch systems and to be sufficiently flexible to adapt to special lunar-landing and lunar-launch techniques.

3.2 PROJECT IMPLEMENTATION CRITERIA.- Considerations which determine the design of the Apollo Spacecraft, its operation, and ground support activity are presented in this section.

3.2.1 Technical Guidelines.- The Technical Guidelines are the collection of principles to which the basic technical approach of the space vehicle system must be responsive. They are the first-order criteria from which successive design criteria, performance margins, tolerances, and environments are developed.

3.2.1.1 Space Vehicle Concept.-

3.2.1.1.1 Launch Vehicle.- The Saturn C-1 Launch Vehicle shall be the basic Launch Vehicle for early phases, however, other launch vehicles may be used for certain development and/or qualification flights. Subsequent project phases will employ advanced Saturn and NOVA-class Launch vehicles.

3.2.1.1.2 Spacecraft.- The Spacecraft shall be composed of separable modules such that (1) "effective weight" principles can be realized through proper jettisoning of expendable units, and (2) module configurations peculiar to specific missions can be modified without substantial effect upon modules common to general missions. The general features of the Spacecraft are described in the following paragraphs.

3.2.1.1.2.1 Command Module.- The Spacecraft shall include a recoverable Command Module which shall remain essentially unchanged for all Apollo missions.

3.2.1.1.2.1.1 Command Center.- The Command Module shall be the space vehicle command center where there are exercised all crew-initiated control functions. As the command center, this module where practical shall contain the communication, navigation, guidance, control, computing, display equipment, etc., requiring crew mode selection. In addition, other equipment required during nominal and/or emergency landing phases shall be included in the Command Module. As the command center, this module shall include features which allow effective crew participation such as windows with a broad field of view for general observation, landing, rendezvous; equipment arrangements allowing access for maintenance;

and simple, manually-operated functions in lieu of complex automation.

3.2.1.1.2.1.2 Housing.- The Command Module shall house the crew during all mission phases and shall contain those experimental measurements obtained during flight to satisfy mission objectives.

3.2.1.1.2.1.3 Reentry and Landing.- The Command Module shall be the reentry and landing vehicle for both nominal and emergency mission phases. The use of equipment such as ejection seats or personal parachutes is not precluded for certain cases.

3.2.1.1.2.1.4 Ingress and egress.- Ingress and egress hatches to the Command Module shall not be obstructed at any stage of space vehicle countdown, flight, and recovery. Means of egress to free space without decompression of the entire Command Module shall be provided.

3.2.1.1.2.2 Service Module.- The Spacecraft shall include an unmanned Service Module for all missions except specific low altitude super-orbital-velocity reentry tests. This unmanned module shall contain stores and systems which do not require crew maintenance or direct operation, and which are not required by the Command Module after separation from the Service Module. The Service Module shall house all propulsion systems except that required for lunar landing and attitude control during earth-entry. Consideration shall be given to inflight maintenance of equipment in the Service Module by crewmen in extra-spacecraft suits. The Service Module may be modified in accordance with particular mission requirements, but the principal structural load paths, geometric arrangement, and configuration shall remain unchanged for various missions and project phases. It is expected that the Service Module would normally be jettisoned prior to reentry into the earth's atmosphere. The Service Module shall not be recoverable.

3.2.1.1.2.3 Lunar Landing Module.- The Spacecraft shall include a Lunar Landing Module for the lunar landing missions.

3.2.1.1.2.4 Spacecraft Adapter.- The Spacecraft Adapters shall structurally and functionally adapt the Service Module or Lunar Landing Module to the launch vehicle for the non-lunar landing and lunar landing configurations, respectively.

3.2.1.1.2.5 Space Laboratory Module.-The Spacecraft for certain earth-orbital flights may include a non-recoverable Space Laboratory Module in which various special tests may be performed. The Space Laboratory Module shall provide the structural and interface functions of an adapter.

3.2.1.1.2.5.1 Support Systems.-The Space Laboratory Module shall have on board sufficient equipment to satisfy its own requirements, manned and unmanned, without demand upon other Spacecraft equipment.

3.2.1.1.2.5.2 Ingress and Egress.-The Space Laboratory shall have a hatch suitable for ingress and egress to free space and for connection with the Spacecraft.

3.2.1.2 Operational Concept.-

3.2.1.2.1 Mission Profiles.-The Spacecraft shall be designed with the capability of performing a variety of missions including earth orbital, circumlunar, lunar orbit, and lunar landing.

3.2.1.2.2 Manning of Flights.-The Spacecraft shall be designed for manned operation with no system requirement for unmanned missions. Where unmanned development flights are required specially equipped Spacecraft will be used.

3.2.1.2.3 Onboard Command.-The primary command and decision-making responsibility shall be on board the Spacecraft. The Spacecraft shall have the capability to perform the mission independent of ground-based information. This shall not preclude the use of ground-based information for crew use to increase reliability, accuracy, and performance.

3.2.1.2.4 Flight Crew.-The flight crew shall consist of three men.

3.2.1.2.4.1 Crew Participation .-The flight crew shall control or direct the control of the Spacecraft throughout all flight modes. They shall participate in navigation, control, monitoring, computing, repair, maintenance, and scientific observation when advantageous. Status of systems shall be displayed for crew assessment and operational mode selection including Spacecraft and launch-vehicle-systems status, staging sequences, and touchdown control. The Spacecraft shall be designed so that any single crewman will be able to perform all tasks essential to return the Command Module.

3.2.1.2.4.2 Crew Mobility.- The onboard command guideline requires a considerable degree of crew mobility. Towards this end, a "shirtsleeve" environment shall be provided during all flight phases.

3.2.1.2.4.3 Automatic Systems.- Automatic systems shall be employed to obtain precision, speed of response, or to relieve the crew of tedious tasks; but crew monitoring of these systems with provisions for crew override or mode selection is required.

3.2.1.2.4.4 Abort Initiation.- Initiation of abort and subsequent control of abort modes shall be primarily the responsibility of the crew. There shall be no abort responsibility assigned to ground command or automatic systems except during prelaunch and launch periods if there is insufficient time for crew action. In such event, abort may be initiated without crew cognizance but subsequent flight control shall be the responsibility of the crew. Automatic and manual abort sequence modes shall be available for crew selection.

3.2.1.2.5 Flight-Time Capability.-

3.2.1.2.5.1 Flight Period.- The Spacecraft systems shall be capable of performing at their nominal design performance level for a mission of 14 days without resupply. For lunar-landing missions 7 of the 14 days may be on the lunar surface.

3.2.1.2.5.2 Postflight Period.- The Command Module shall provide the crew a habitable environment for one day and a survivable environment for one week following a land or water landing.

3.2.1.2.6 Landing.- The Spacecraft shall have the capability of initiating a reentry and landing maneuver at any time during either lunar or orbital missions. Prior to each flight, a primary ground landing site and suitable backup landing site will be selected for normal mission landing. Additional criteria apply as follows:

3.2.1.2.6.1 Lunar Missions.- Alternate landing sites shall be designated prior to flight such that a landing is possible at these sites regardless of the time of reentry.

3.2.1.2.6.2 Earth-Orbital Mission.- The Spacecraft shall be capable of landing at the primary landing site (or at the backup site) from at least three orbits per day. In addition, alternate sites which may involve either land or water landing will be designated such that at least one alternate site can be reached for a landing from each orbit.

3.2.1.2.7 Ground Monitoring and Communication.-

3.2.1.2.7.1 Earth-Orbital Missions.-

3.2.1.2.7.1.1 Monitoring.-

3.2.1.2.7.1.1.1 Powered Flight.- There shall be continuous monitoring of onboard system and crew status during powered flight.

3.2.1.2.7.1.1.2 Orbital Flight.- Flight progress, onboard systems operation, and crew status shall be monitored by the ground operational Support System, a minimum of one contact with the Spacecraft per hour.

3.2.1.2.7.1.2 Ground Communications.- The network shall operate on a centralized control basis.

3.2.1.2.7.2 Lunar Missions.- Communications and ground tracking shall be provided throughout the lunar mission for the period between leaving the earth parking orbit and the initiation of earth reentry except where limited by the Spacecraft being blanketed by the moon or where it is mutually agreed by MSC and the contractor that such coverage is infeasible.

3.2.1.2.8 Apollo Control Center.- All phases of Apollo missions shall be directed from an Apollo Control Center.

3.2.1.2.9 Communications Center.- All mission communications during Apollo missions shall be controlled by the Apollo Control Center.

3.2.1.2.10 Tracking and Ground Instrumentation Network.- All existing networks and associated facilities shall be considered for support of an Apollo mission where practical.

3.2.1.3 Reliability and Crew Safety.-Mission reliability and crew safety goals, assuming a launch vehicle reliability of 0.95 and including the effect of ground complex reliability, but excluding consideration of radiation and meteoroid impact, shall be as follows:

3.2.1.3.1 Mission Reliability.-The probability of accomplishing the mission objectives shall be 0.90.

3.2.1.3.2 Crew Safety.-

3.2.1.3.2.1 Nominal.-The probability that none of the crewmen shall have been subjected to conditions greater than the nominal limits specified in Design Criteria, shall be 0.90.

3.2.1.3.2.2 Emergency.-The probability that none of the crewmen shall have been subjected to conditions greater than the emergency limits specified in Design Criteria shall be 0.999.

3.2.2 Design Criteria.-Design and operational procedures shall be conducted in accordance with rational design principles.

3.2.2.1 Limit Conditions.-The design limit load envelopes shall be established by superposition of rationally deduced critical loads for all flight modes. Load envelopes shall recognize the cumulative effects of additive-type loads. No system shall be designed incapable of functioning at limit load conditions.

3.2.2.2 Spacecraft Maintenance.-Equipment arrangements, accessibility, and interchangeability features that allow efficient preflight and inflight servicing and maintenance shall be given full consideration. The use of automatic checkout equipment shall not preclude manual entrance into and checkout of the system being checked. Design considerations shall also include efficient mission scrub and recycle procedures.

3.2.2.3 Ground Handling.-Full design recognition shall be given to the durability requirements of Spacecraft equipment and systems subjected to the continuous handling and "wear-and-tear" of preflight preparation.

3.2.2.4 Command Module Reuse.-The Command Module and internal systems shall be designed for repeated mission reuse after recovery. The internal systems shall be designed for an operational life of three nominal missions, exceptions are made for systems and components normally classified as expendable and those flight items which would be unduly compromised in design by environmental conditions occurring after their operational function has been performed.

3.2.2.5 Spacecraft Water Stability.-Spacecraft flotation and water stability characteristics shall be such as to ensure that the Spacecraft will recover from any initial attitude and will float upright with normal egress hatches clear of the water. Spacecraft seakeeping capability shall be such as to ensure a 7-day flotation period.

3.2.3 Performance Criteria.- Rational margins shall be apportioned to systems and components such that the greatest overall design efficiency is achieved within the Launch Vehicle capabilities and implementation criteria constraints. The following specific systems margins are derived from rational consideration of past and anticipated operational experience. They are to be used as design criteria until experience justifies modification.

3.2.3.1 Multiple Failure.- The decision to design for single or multiple failures shall be based on the expected frequency of occurrance as it affects system reliability and safety.

3.2.3.2 Fail Safe.- System or component failure shall not propogate sequentially, i.e., design shall "fail safe."

3.2.3.3 Design Margins.- All Spacecraft systems shall be designed to positive margins of safety.

3.2.3.4 Repressurization.- The repressurization system shall be designed for two complete cabin repressurizations, a minimum of 18 airlock operations, and a continuous leak rate as high as 0.2 lbs. per hour. Provisions shall be made for recharging portable life support systems ("back packs").

3.2.3.5 Vacuum Operation of Cabin Equipment.- Equipment which is normally operated in the pressurized cabin environment shall be designed to function for a minimum of four days in vacuum without failure.

3.2.3.6 Thermal Resistance.- The Spacecraft modules shall be designed such that additional or lesser requirements in thermal resistance may be accommodated or taken advantage of without major overall design changes.

3.2.3.7 Meteroid Protection.- The Spacecraft modules shall be designed such that additional or lesser requirements in meteroid protection may be accommodated or taken advantage of without major overall design changes.

3.2.3.8 Radiation Shielding.- The Spacecraft modules shall be designed such that additional or lesser requirements in radiation protection may be accommodated or taken advantage of without major overall design changes.

3.2.3.9 Isolation of Modifications.- The Spacecraft modules shall be designed such that general modifications to one module do not propagate through the other modules.

3.2.3.10 Advances in Technology.- Flexibility shall be incorporated into the design such that advantage can be taken of advances in technology.

3.2.3.11 Off-the-Pad Capability.- The Launch Escape Propulsion System shall be capable of lifting the Command Module off the launch vehicle on the pad to an altitude of 5000 feet and a lateral range at touchdown of at least 3000 feet.

3.2.3.12 Special Flight Loads.-

3.2.3.12.1 Tumbling at Maximum Dynamic Pressure.- Primary structures are to be designed for loads arising from a "tumbling" of the escape vehicle at maximum dynamic pressure during launch.

3.2.3.12.2 20g Reentry.- Primary structures are to be designed for a limit load of 20g during reentry.

3.2.3.12.3 Noise.- The design shall accommodate sound pressure levels of 166 db in the frequency range 4 to 9600 cps emanating from the Launch Escape Propulsion System during both launch and abort modes.

3.2.3.12.4 Buffet.- The design shall accommodate a buffet pressure of 1.5 psi (rms) in the frequency range of 0 to 4 cps on the Service Module and Adapter during the earth-launch phases.

3.2.3.13 Structural Design Factors.-

3.2.3.13.1 Ultimate Factor.- The ultimate factor shall be 1.5 applied to limit loads. This factor may be reduced to 1.35 for special cases upon rational analysis and negotiation with Manned Spacecraft Center.

3.2.3.13.2 Pressure Vessel Design Factors.- Pressure vessels are to to be designed using the following factors based on limit loads.

3.2.3.13.2.1 Pressure Vessel Proof Factor.- The proof factor shall be 1.33 when pressure is applied as a singular load. This factor may be reduced for special cases upon rational analysis and negotiation with Manned Spacecraft Center.

3.2.3.13.2.2 <u>Pressure Vessel Ultimate Factors</u>.- The ultimate factor shall be 2.00 when pressure is applied as a singular load. This factor may be reduced to 1.5 for special cases upon rational analysis and negotiation with Manned Spacecraft Center. The main propellant tanks are a special case and will have an ultimate factor of 1.5.

3.2.3.13.3 <u>Pressure Vessel Limit Loads</u>.- Limit loads shall be obtained with limit pressures. When pressure effects are relieving, pressure should not be used.

3.2.3.13.4 <u>Pressure Stabilized Structures</u>.- No primary structures shall require pressure stabilization.

3.2.4 Nomenclature.-

3.2.4.1 Reference Axes.- The reference axes of the Spacecraft shall be orthogonal and identified as shown in figure 1.

3.2.4.1.1 X-Axis.- The X-Axis shall be parallel to the nominal launch axis of the Space Vehicle and be positive in the direction of initial flight.

3.2.4.1.2 Y-Axis.- The Y-Axis shall be normal to the X-Axis and positive to the right of a crewman when the crewman is facing towards positive X.

3.2.4.1.3 Z-Axis.- The Z-Axis shall be normal to both the X and Y axes and be positive in the direction of the crewman's feet.

3.2.5 Crew Requirements.- Design and operational procedures shall be in accordance with the crew requirements data presented here. The data presented are for various limits as defined below.

3.2.5.1 Nominal Limits.- Nominal limits are defined as the limits within which the crew's environment shall be maintained during normal operations.

3.2.5.1.1 Nonstressed Limits.- Nonstressed limits are defined as the environmental limits to which the crew may be subjected for extended periods of time such as orbit, lunar transit, and periods subsequent to normal landings.

3.2.5.1.2 Emergency Limits.- Emergency limits are defined as the environmental limits beyond which there is a high probability of permanent injury, death, or incapacity to such extent that the crew could not perform well enough to survive.

3.2.5.2 Metabolic Requirements.- The average daily metabolic requirements for each crew member are listed below.

Oxygen consumption	1.8 lb/day/man
Carbon dioxide output	2.3 lb/day/man
Heat output	11,300 BTU/day/man
Water consumption	6.0 lb/day/man (This includes water in food; additional water may be required for sanitation.)
Food consumption	2800 Kcal/day/man

3.2.5.3 Crew Environment Requirements.-

3.2.5.3.1 Cabin Pressure.- The cabin pressure nominal limits shall be 3.5 psia minimum and 15.0 psia maximum. The emergency limit shall be 3.5 psia minimum.

3.2.5.3.2 Oxygen Partial Pressure.- The oxygen partial pressure nominal and emergency limits shall be 160 mm Hg minimum.

3.2.5.3.3 Carbon Dioxide Partial Pressure.- The carbon dioxide partial pressure nominal limit shall be 7.6 mm Hg maximum. The emergency limits are presented in figure 2.

3.2.5.3.4 Cabin Temperature.- The cabin temperature nonstressed limits shall be 70° F minimum and 80° F maximum. The stressed and emergency limits are presented in figures 3 and 4, respectively.

3.2.5.3.5 Cabin Relative Humidity.- The cabin relative humidity nonstressed limits shall be 40 percent minimum and 70 percent maximum. The stressed and emergency limits are presented in figures 3 and 4, respectively.

3.2.5.3.6 Radiation.- The nominal limit shall be the average yearly exposure tabulated in figure 5. The emergency dose limits shall be the maximum permissible, single acute emergency dose as tabulated in figure 5. Dosage calculations shall be based on the model presentation in figure 6. In the absence of sufficient information to assign dose value due to secondary radiation, a value of 50 percent of the primary dose will be used.

3.2.5.3.7 Noise.- The noise nonstressed limit shall be 80 db over-all and 55 db in the 600 cps to 4800 cps range. The stressed limit shall be the maximum noise level which will permit communications with the ground and between crew members at all times. The emergency limit is presented in figure 7.

3.2.5.3.8 Vibration.- The vibration stressed, nonstressed, and emergency limits are presented in figure 8.

3.2.5.3.9 Sustained Acceleration.- The sustained acceleration limits for eyeballs out, down, and in conditions are presented in figures 9, 10, and 11. The limits presented were obtained from references 1 through 8, and are for currently available restraint systems, optimum body positioning, and without the use of G-suits. The sustained acceleration performance limits are defined as the maximum sustained acceleration to which the crew shall be subjected and still be required to make decisions, perform hand controller tasks requiring visual acuity, etc.

3.2.5.3.10 Impact Acceleration.- The impact acceleration nominal and emergency limits are presented in figures 12 and 13, respectively.

3.2.6 <u>Natural Environment</u>.- Design and operational procedures shall be in accordance with the natural-environment data presented here. It should be recognized that all natural environment data required for the project are not included herein.

3.2.6.1 <u>Launch</u>.-

3.2.6.1.1 <u>Atmospheric Pressure, Density, and Temperature</u>.- The surface variation of atmospheric pressure, density, and temperature is given in reference 9.

3.2.6.1.2 <u>Wind</u>.- The direction, magnitude, and cumulative percentage of surface winds are given in reference 10.

3.2.6.1.3 <u>Precipitation</u>.- The average monthly precipitation is given in figure 14.

3.2.6.1.4 <u>Thunderstorms</u>. - The average number of hours per month during which there are thunderstorms is shown in figure 15.

3.2.6.1.5 <u>Surface Temperature</u>.- The maximum, minimum, and average temperatures are given in figure 16.

3.2.6.2 <u>Flight</u>.-

3.2.6.2.1 <u>Atmospheric Phase</u>.-

3.2.6.2.1.1 <u>Atmospheric Pressure, Density, and Temperature</u>.- The altitude variation of atmospheric pressure, density, and temperature is given in reference 9.

3.2.6.2.1.2 <u>Wind</u>.- The variation of wind with altitude is given in reference 10.

3.2.6.2.2 <u>Mission Phase</u>.-

3.2.6.2.2.1 <u>Solar Phenomena</u>.- The hazards associated with an active sun are presented as a model solar event system with an indicated average frequency of occurance.

3.2.6.2.2.1.1 <u>Model Solar Event</u>.- The time integrated model solar event is shown in figure 17. In arriving at this spectrum it is assumed that the flux of particles in energy range E to E + dE can be described by

$$\phi(E) = kt \qquad t < t_o$$
$$\phi(E) = k\left(\frac{t}{t_o}\right)^2 \qquad t > t_o$$

t_o (time of maximum flux) is related to energy by

$$E = e^{-.13t}$$

3.2.6.2.2.1.2 Probability of Encounter.- The probability of encounter of a solar event shall be assessed on the basis of total particles in the event. Figure 18 presents the average frequency for particles above 100 MEV over a seven-day mission. The total number of particles between 5 to 100 MEV range is shown in figure 17.

3.2.6.2.2.2 Van Allen Radiation Belts.- A description of the Van Allen radiation belts is presented in figure 19.

3.2.6.2.2.2.1 Inner Belt.- The inner belt is concentrated between the geomagnetic latitudes of 25 degrees North and 25 degrees South. It initiates as an altitude of 500 km and peaks in intensity at an altitude of 8500 km. The proton spectrum at the geomagnetic equator is presented in figure 20.

3.2.6.2.2.2.2 Outer Belt.- The outer belt is concentrated between the geomagnetic latitudes 50 degrees North and 50 degrees South. It initiates at an altitude of 15,000 km, peaks in intensity at an altitude of 16,000 km, and decreases to a minimum intensity at an altitude of 21,000 km. The distribution of particles in the heart of the outer belt is presented in figure 21.

3.2.6.2.2.3 Meteroid Considerations.- The hazards involved in encountering meteroid will be assessed on sporadic activity only. The flux considerations for sporadic activity shall be based upon the Whipple distribution presented in figure 23.

3.2.6.2.2.4 Electromagnetic Radiation.- Electromagnetic radiation to be used for Spacecraft environmental analysis is presented in reference to its source.

3.2.6.2.2.4.1 Solar Radiation.- The electromagnetic radiation from the sun covering the spectrum from 60 angstroms to 1300 angstroms is given in figure 24, from 1300 angstroms to 2000 angstroms is given in figure 25 and from .2 microns to 2.0 microns is given in figure 26.

3.2.6.2.2.4.2 Earth Radiation and Reflection.- The earth's albedo shall be considered as 35 percent. The remaining 65 percent shall be considered to be absorbed and some re-emitted as thermal radiation. The spectrum for the earth's albedo at local noon is given in figure 27. The radiation at the center of the dark side shall be considered to originate from a 251° K black body.

3.2.6.2.2.4.3 Lunar Surface Properties.- The physical characteristics of the lunar surface and topography are given in figure 28. A representative cross-section of the lunar surface is shown in figure 29.

3.2.6.2.2.4.4 Background Radiation.- The background radiation from celestial sources shall be considered to be 10^{-4} ergs/cm^2sec in the interval 1230 to 1350 angstroms.

3.2.6.2.2.5 Interplanetary Atmosphere.- The interplanetary atmosphere shall be considered as shown in figure 30.

3.2.6.2.2.6 Space Background.- The space background electromagnetic radiation is presented above. The corpuscular radiation shall be considered as shown in figure 31 which represents the cosmic ray flux.

3.2.6.2.2.7 Earth Gravitational and Geometrical Constants.- The following earth gravitational and geometrical constants are to be used for tracking and orbital computations:

3.2.6.2.2.7.1 Symbols

a equatorial radius, meters

E oblateness factor = $\left(1 - \frac{\text{minor diameter}}{\text{major diameter}}\right)$

g acceleration of gravity at equator, meters/sec^2

G universal gravitational constant

h altitude above the reference ellipsoid, meters

J	harmonic terms of the potential function
M	mass
P_n	$(\sin\phi)$ Legendre polynomial
ϕ	latitude
r	radius from center of earth, meters
u, v, w	axis system ordinates, meters
U	potential function
ϕ	longitude
ω_e	rotational speed of earth $\dfrac{2\pi}{8.6.64.0982 + .00164T}$
T	Julian centuries (36525 days) from 1900 Jan 0.5 U.T.
A_u	astronomical constant

subscripts

e	earth
s	sun
p	planet
m	moon

3.2.6.2.2.7.2 Gravitational.-

3.2.6.2.2.7.2.1 Numerical Values.- In the formula

$$U = (Gm_e/r)\quad 1 - \sum_M J_M\ (Ae/r)^M P_N\ (\sin\phi)$$

Where P (sin ϕ) is the Legendre polynomial and ϕ is the geocentric latitude, or in alternate notation:

$$(f,\phi) = \frac{GM}{r} e\left[1 + \frac{J}{3}\left(\frac{ae}{r}\right)^2 (1-3\sin^2\phi) + \frac{H}{5}\left(\frac{ae}{r}\right)^3 (3-5\sin^2\phi)\sin\phi + \frac{D}{35}\left(\frac{a_e}{r}\right)^4 (3-30\sin^2\phi + 35\sin^4\phi)\right]$$

$$GM_e = 3.986032\ (\pm 0.000030) \times 10^{14}\ \frac{\text{meters}^3}{\text{sec}^2}$$

$$J_2 = 1082.30\ (\pm 0.2) \times 10^{-6}$$
$$J_3 = -2.3\ (\pm 0.1) \times 10^{-6}$$
$$J_4 = -1.8\ (\pm 0.2) \times 10^{-6}$$
$$J_n = 0.0\ (\pm < 1.0) \times 10^{-6} \quad n \geq 5$$
$$J_{nm} = 0.0\ (\pm < 2.0) \times 10^{-6},\ m \neq 0$$
$$a_e = 6.378165\ (\pm 0.000025) \times 10^6 \text{ meters}$$
$$J = 1.62345 \times 10^{-3}$$
$$H = -0.575 \times 10^{-5}$$
$$D = .7875 \times 10^{-5}$$

3.2.6.2.2.7.2.2 Remarks

3.2.6.2.2.7.2.2.1 The values of GM_e. H, D and a_e are consistent with the values of geodetic parameters.

$1/\varepsilon$ = Reciprocal of earth/flattening = 298.30

g_e = 978.030 cm/sec^2

3.2.6.2.2.7.2.2.2 The values of a_e, e, g_e are those specified in the DOD World Geodetic System 1960 and are here recommended for the sake of consistency. In addition, they are close to the best estimates for these parameters. Reasonable alternative values based on terrestrial geodetic data; e.g., those in reference 11 differ by less than 20 meters is a_e, .00001 meters^2 in g_e, and 0.1 in 1/f.

3.2.6.2.2.7.2.2.3 The value of g_e incorporates a correction of .0013 meters/sec^2 to the Potsdam standard absolute gravity.

3.2.6.2.2.7.2.2.4 The values of J_3 and J_4 are compromises between the values obtained by the principal investigators of satellite orbits as presented in references 4, 5, and 6, with greatest weight to reference 6 and the given uncertainties are based on the discrepancies between these results. The values of J_2 by these same investigators range from 1082.19 to 1082.79×10^{-6}. The magnitude of effect of the omitted J_{nm} on satellite positions is about ±400 m or less (see reference 12).

3.2.6.2.2.7.2.2.5 The most serious discrepancy of determination of gravitational parameters is between the GM from terrestrial data, 3.1986032 (±0.000030) × 10^{14} meters^3/sec^2, and that based on the lunar mean motion and the radar measurement of the moon's distance: 3.986141 (±0.000040) × 10^{14} meters^3/sec^2. This value depends on the moon/earth mass ratio of 1/81.375 (see reference 8); 3.986048 is obtained from Delano's 1/81.219 (see reference 13). However, the stated uncertainty depends mainly on the uncertainties in the radar measurement and the lunar radius.

3.2.6.2.2.7.3 <u>Geometrical</u>.-

3.2.6.2.2.7.3.1 <u>Numerical Values</u>.- Figure 32 represents the astro-geodetic geoid data station spacing and distribution. The Coordinate System used has its u, v, and w axes earth-centered, earth-fixed, and directed toward the latitudes and longitudes 0°, 0°; 0°, 90° E; and 90° N., respectively.

3.2.6.2.2.7.3.1.1 <u>Corrections - General</u>.- The corrections to be added to the rectangular coordinated in the u, v, and w system are presented in figure 33. These corrections are based on reference 11.

3.2.6.2.2.7.3.1.2 <u>Corrections at Stations Not Connected to the Geodetic System</u>.- Stations not connected to any of the principal Geodetic Systems, but which have an astronomic position or which are connected to a local system must be treated in the following way. The geodetic latitude and longitude is to be that of the astronomic or local system, and the u, v, and w coordinated obtained by the equations.

$$u = (\gamma + h) \cos \phi \cos \lambda$$
$$v = (\gamma + h) \cos \phi \sin \lambda$$
$$w = \left[(1 - e^2)\, \gamma + h\right] \sin \phi$$

where

$$\gamma = a_e\, (1 - 3^2 \sin \phi)^{-1/2}$$
$$e = 2f - f^2$$

and h is the elevation above the ellipsoid. If $a_e = 6378165$ meters and $f = 1/298.30$ are used, and the height above the ellipsoid assumed to be identical with the height above sea level, then the standard error of position in the radial direction should be

$$\sigma(r) = \pm 45 \text{ meters.}$$

If the geoid heights from figures 2 or 3 of reference 3 are added to the height above sea level, then there will be a slight improvement to about $\pm$ 35 meters.

For the horizontal coordinates at a station in a geophysically stable continental area.

$$\sigma(r\ \phi) = \sigma(r\ \lambda \cos \phi) \approx \pm 170 \text{ meters.}$$

For the horizontal coordinates from a single astronomic position on an island or in a geophysically disturbed area (mountains, etc.)

$$\sigma(r\ \phi) = \sigma r\ \lambda \cos \phi) \approx +350 \text{ meters.}$$

By using the mean position obtained by connecting astronomic observations on opposite sides of an island by traverse this may be improved to about

$$\sigma(r\ \phi) = \sigma r\ \lambda \cos \phi) \approx + 250 \text{ meters.}$$

By using topographic isostatic corrections of the deflections of the vertical this may further be improved to about

$\sigma(r\ \phi) = \sigma(r\ \lambda \cos \phi) \approx \pm 200$ meters (for a single station) and $\sigma(r) = \sigma(r\ \bar{\lambda} \cos \phi) \approx \pm 120$ meters (for the mean from observations on opposite sides.)

3.2.6.2.2.7.3.2 Remarks -

The values recommended for Argentina and Australia are based on the assumption of tangency at the geodetic datum "origins" of an a_e = 6378165+N_o, 1/298.3 ellipsoid, where N_o is the geoid height at the datum origin given in figures 2 and 3 of reference 11.

The Vanguard Datum was based on the assumption of tangency to NAD at its origin (97°N, 263°E) of the Hough Ellipsoid

a_e = 6378270

f = 1/297.0

The SAO SP 59 datum (see reference 14) is based on the assumption of tangency to the conventional datums, corrected by gravimetrically computed deflections, of the vertical (except in Argentina), of the International Ellipsoid

a_e = 6378388

f = 1/297.0

The large differences from reference 11 datum are due mainly to this use of an obsolete ellipsoid and secondarily to the utilization of much less observational data.

Note that all datum shifts are described as translations; there are no rotations. For properly observed geodetic systems, the orientation error is negligible. Orientation of geodetic systems is obtained from the stars through "Laplace stations," at which astronomic azimuth and longitude are observed.

The standard error for difference of position between two stations connected to the same geodetic control system should always be less than $\pm$ 20 meters.

The standard errors for astronomic positions in a continental area is based on autocovariance analysis of gravimetry.

The standard error for astronomic positions on islands is based on a sample of 69 islands in the West Indies connected to the continental geodetic system by Hiran trilateration.

3.2.6.2.2.8 Sun and Planetary Constants.- Certain sun, lunar, and planetary constants to be used are presented in figure 34.

3.2.6.2.3 Entry, Landing, and Recovery Phase.-

3.2.6.2.3.1 Atmospheric Pressure, Density, and Temperature.- The altitude, seasonal, daily and latitude variation of pressure, density, and temperature will be as presented by the revised ICAO reference atmosphere. Because this reference is in the process of publication the 1959 ARDC standard atmosphere will be used until the ICAO data is available.

3.2.6.2.3.2 Wind Velocity.- The wind velocity which is exceeded only 10 percent of the time is presented in figure 35 for the months of July and January.

3.2.6.2.3.3 Wave Height.- The wave height which is exceeded only 10 percent of the time is presented in figure 36 for the months of January and July.

3.3 FLIGHT PLAN.-

3.3.1 General.- Definitive flight plans for the various missions which will be included in the Apollo program can be formulated only after the design details of the Space Vehicle are better known, mission objectives defined in more detail, and more comprehensive information is available concerning the tradeoffs between the variables which describe each mission phase. There are, however, some requirements concerning flight plans and some information relative to the choice of trajectories which are presently known. The following sections present the preliminary requirements for flight plans, the effect of variations of certain mission parameters upon flight plans, and an example of a lunar landing mission trajectory.

3.3.2 Preliminary Flight Plan Requirements.- The following flight plan requirements are considered to be preliminary and a partial list. Further studies and experience in formulating flight plan and Spacecraft design details on the part of NASA and the contractor will enable more definitive requirements to be specified.

a. Launch Site - All earth-orbit and lunar missions are to be launched from Cape Canaveral, Florida. This does not preclude the use of other launch sites for systems tests. The launch azimuths are to be within the limitations set by range safety and tracking considerations.

b. Launch Time Window - Lunar mission flight plans must include at least a 2-hour period on launch date in which the mission can be launched either continuously or at discrete intervals.

c. Number of Parking Orbits - Multiple parking orbits are acceptable but are not necessary.

d. Earth Orbits - Earth-orbit altitudes for manned orbital flight and parking orbits for lunar missions will be limited to altitudes from 90 to 400 nautical miles.

e. Translunar Insertion Position - Final insertion into the translunar trajectory shall be located such that the trajectory can be determined by the Ground Operational Support System within 15 minutes of translunar insertion burnout.

f. Translunar Midcourse Corrections - As a design objective the $3\ \sigma$ velocity requirement for midcourse navigation shall not exceed 500 ft/sec including the error in arrival velocity.

g. Lunar Landing - The lunar landing shall be initiated from a lunar orbit. The nominal orbit altitude is 100 nautical miles. The landing techniques are to use elliptical transfer to lower altitudes which do not intersect the moon's surface. A 5° plane change capability shall be supplied for establishing the initial orbit.

h. Landing Site - Mission plans may call for several landing sites. The following factors will be considered in the choice of a landing site:

(1) Propulsion and fuel requirements.

(2) Maneuvering and hovering capability.

(3) Communication with GOSS.

(4) Illumination.

(5) Temperature of environment.

(6) Surface texture.

(7) Ease of identification.

i. Lunar Launch - There shall not normally be a requirement to reposition the spacecraft attitude prior to lunar launch.

j. Transearth and Midcourse Corrections - As a design objective the $3\ \sigma$ velocity requirement for midcourse navigation shall not exceed 500 ft/sec. The inclination of the transearth trajectory to the earth's equator shall be compatible with existing tracking stations.

k. Reentry - The reentry vehicle shall be capable of reentry over a nominal 30, nautical mile corridor with peak deceleration limited to 10g. The direction of reentry to be with the rotation of the earth.

3.3.3 Trajectory Characteristics.- The material is presented in the order of the different phases of flight.

a. Launch-Time Window - In order to provide a launch-time window it is necessary either to maneuver the Launch Vehicle or Spacecraft to intercept a planned nominal trajectory, or to select a new trajectory which will satisfy the mission objectives and which can be obtained at the actual launch time. Both the lunar trajectory selection and maneuvering of the Launch Vehicle methods of obtaining a launch window should be developed for use in the Apollo mission. The discussion of launch window below is limited to lunar trajectory variation because of the lack of detail at present on the booster for use in lunar missions.

   To a first order of approximation, the Spacecraft can be injected into a lunar trajectory from any parking orbit which passes over the earth surface point which is formed by projecting the line of centers between the earth and the moon at the time of closest initial approach. With no restriction due to the mission objectives or performance loss, a launch could be made at any time of day. The launch window is, therefore, primarily a function of the permissible azimuth swing for launch from Cape Canaveral. Figure 37 shows the launch-time window and the maximum inclination of the parking orbit as a function of azimuth variations, positive and negative from due East launches from Cape Canaveral. This launch window is independent of lunar declination and can be obtained for lunar injection toward either the South or North.

b. Parking Orbits - Earth-orbit altitude from 90 to 400 N.M. may be required for earth-orbit checkout and rendezvous. For direct lunar missions, however, it is expected that the best launch booster performance is obtained with low altitude parking orbits. A nominal value of orbit altitude for direct lunar missions is 600,000 feet. The effect of launch delays on the earth track of the parking orbit and the location of the injection point is shown in figure 38. The extreme orbit paths for a 4-hour launch window are shown as the outside solid lines for the condition where the launch azimuth variation is symmetrical, about a due East launch from Cape Canaveral. The inner broken lines show the extremes for a 2-hour launch window. The launch window results from the use of any trajectory between these extremes. The location of the injection points for some important lunar declination is shown in figure 38.

c. Injection - The characteristic velocity requirement for injection into translunar trajectories from a 600,000 foot parking orbit is shown in figure 39 for initial engine thrust to Spacecraft weight ratios from 0.5 to 1.5. Their results are applicable for a specific impulse from 250 to 500. Even though the results shown are terminated at 900,000 feet, it is apparent that with propulsion units with initial thrust to weight ratios less than 1.0, the injection is possible at 200 N.M. altitude without significant losses in performance.

The approximate area covered by the Mercury tracking network for 5° elevation is shown in figure 40 for several altitudes. Comparing figures 38 and 40, it is apparent that many of the Mercury tracking stations are poorly located for coverage at injection of a 4-hour launch window. For maximum Northern declination of the moon, the injection point can be tracked for nearly 4 hours of launch window with the station in Australia. With the moon at maximum Southern declination, the injection point can be covered for about 3 hours of launch window with the tracking stations located in the USA. At lunar declination near 0°, however, the launch-time window has restricted coverage. Relocation of some stations and addition of several more would be required to give adequate coverage at all declinations.

[redacted]

d. Translunar Trajectory Characteristics - The nominal translunar trajectory for all lunar missions is one which has a coast return to the earth with acceptable reentry conditions. For circumlunar missions this trajectory must have a flight time and return inclination which returns the vehicle so that the primary landing area is within the reentry maneuver capability of the reentry vehicle. The translunar trajectories for lunar landing missions approaches the moon to within 100 nautical miles altitude in order to minimize the landing propulsion requirements.

The inclination of the translunar trajectory plane is a function of the parking orbit inclination and the lunar declination as shown in figure 41. Varying the inclination of the parking orbit to obtain launch time tolerance as indicated in this section will result in a change in the translunar trajectory inclination which in turn will have some effect on the inclination of the lunar orbit unless plane changes are made during transfer.

The translunar trajectory is tracked with the deep-space network. Figure 42 shows the coverage of the existing deep-space network at various altitudes. About 15 minutes after injection, the translunar trajectory will be at high enough altitude for tracking.

e. Lunar Orbit - Flight plans require establishing a circular lunar orbit at 100 nautical miles altitude for lunar landing missions. For lunar orbit missions flight plans may call for both circular and elliptical orbits within the limits of propulsion requirements for the 100 nautical mile circular orbit. Velocity increments for establishing lunar orbits are shown in figure 43 (note, add 200 ft/sec for out-of-plane requirements).

f. Lunar Landings - A technique for lunar landing is illustrated in figure 44. The Spacecraft arrives behind the moon on a circumlunar trajectory, a transfer is made to a 100 N.M. circular orbit about the moon. The Spacecraft passes over the landing area once and then at the proper position in the orbit a transfer is made to an elliptical orbit

[redacted]

having a pericynthion of 50,000 feet. The landing run is initiated at 50,000 feet altitude. The impulse requirements for landing on the moon for an optimum flight path for elliptical orbits having 100 nautical miles apocynthion and various pericynthions is shown in figure 45 for various initial thrust-to-weight ratios. Results are shown for termination of the landing run either horizontally or vertically at altitudes below 1000 feet.

g. Lunar Landing Site - Figure 46 shows the lunar landing area available for missions with translunar trajectories which return to earth with posigrade reentry. The available landing area without orbit transfer is limited to a band of latitude approximately + 10 degrees. The landing area is further restricted by the relation of the velocity at injection to the earth-moon plane. In general, the areas labled N are available for injection with velocity directed toward the North and the areas labled S with velocity vectors directed toward the South. Referring this information back to the parking orbit about the earth, it is observed that on the first pass injection toward the North will occur between longitude -100 to +90, which is generally over the Pacific Ocean. It is assumed that the first half of the parking orbit may be needed for systems check so that injection toward the South would be on the first half of the second orbit, roughly between -90 and +65 degrees longitude.

h. Lunar Launch - The probable technique for launch from the moon on the return trip to earth is to lift off in an essentially vertical manuever from the local surface and program pitch into an elliptical orbit. The orbit is circularized at apocynthion and injection on the transearth trajectory is made at the proper time. Figure 47 shows the characteristic velocity requirements for vertical launch into elliptical parking orbits.

i. Transearth- The inclination and the time of flight of the transearth trajectory are used to control the reentry in such a way that the reentry track will be over existing network facilities and traverses reasonable recovery areas. Inclination appears to be in the range of from 30° to 35° which makes use of existing facilities and is compatible with landing site in Southern Texas, Hawaii, and Australia. The injection requirements for transfer from lunar orbit to the transearth trajectory are nearly the same as those for establishing lunar orbits shown in figure 43. The nominal reentry for Apollo missions is to be with posigrade motion with respect to the earth to reduce the reentry heating and widen the reentry corridor.

j. Reentry - The 10g reentry corridor for L/D=0.5 is 40 N.M. for an ARDC-1959 standard atmospheres. The effects of atmospheric variation reduce the reentry corridor to about 35 N.M. Losses due to reentry control techniques which do not use negative lift amount to about 5 N.M. The maximum reentry corridor for Apollo missions for 10g maximum deceleration is 30 N.M.

The location of the reentry point is determined by the declination of the moon at the time the transearth trajectory is initiated, the transit time, and the inclination of the return orbit.

The locus of reentry points for a landing site in Southern Texas and in Australia is shown in figure 48 for several lunar declinations. The track of a 30° reentry orbit indicates that a range after reentry of 7,000 to 8,000 miles is required to return to the Southern Texas for reentry at all lunar declinations. The use of a second landing site in Australia would reduce the maximum required reentry range to about 5,300 nautical miles. Return would be to Texas for Southern declination of the moon and to Australia for Northern declinations of the moon.

A typical reentry from a lunar mission for landing in Southern Texas is shown in figure 48 along with the landing areas, possibly with an L/D = 0.5 vehicle.

3.3.4 Example Flight Plan.- The flight plan presented is an example and does not represent design criteria.

a. Lift-Off Conditions - The launch azimuth is 91°.

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b. Lift-Off to Parking Orbit - The characteristics of the flight plan from launch to insertion into a parking orbit at an altitude of 600,000 feet are presented in figures 49a and 49b.

c. Parking Orbit - Ground tracks for the initial earth orbit having a launch azimuth of 91° are presented in figure 50. The parking orbit is circular at an altitude of 600,000 feet. The only portion of the Continental United States over which the Spacecraft passes during the first revolution is Southern Texas.

d. Parking Orbit to Translunar - The location of the beginning of the insertion phase may be anywhere along the parking orbit depending upon the moon's declination. Figure 50 shows earth tracks for an insertion location in the mid-Pacific region. The characteristics of the flight plan from the parking orbit to the translunar trajectory are presented in figure 51.

e. Translunar and Transearth - Figure 52 presents the translunar and transearth trajectories for the inertial earth-moon system. The translunar trajectory has the characteristic that if no velocity increment is applied, the Spacecraft will return to earth at acceptable reentry conditions. The pericynthion altitude at the moon is 600,000 feet. The return or transearth trajectory shown in figure 53 represents a continuation of the translunar trajectory with a break of 25 hours for landing on the moon and take-off. The transearth and translunar trajectories combined form a reference circumlunar trajectory with proper correction for the lunar time break.

f. Lunar Orbit - The velocity increment required to place the Spacecraft in a 100 nautical-mile circular orbit from the approach pericynthion altitude of 100 nautical miles is seen to be 3025 ft/sec. The landing site is surveyed as the Spacecraft passes over this area during its first revolution. As the Spacecraft approaches a point 180° from the landing site, a velocity impulse of 180 ft/sec is applied to place the Spacecraft in an elliptic orbit with a pericynthion altitude of 50,000 feet. Figure 44 describes the complete technique from translunar trajectory to the lunar landing phase.

g. Lunar Landing - The lunar landing maneuver is initiated at approximately 50,000 feet altitude. The characteristics of the flight plan during this maneuver are presented in figure 54. The maneuver ends at an altitude of 650 feet at which time the Spacecraft vertical and horizontal velocity are near zero. Descent from orbit can be made to any landing site in the orbit plane.

h. Lunar Launch to Transearth - The characteristics of the flight plan from lunar take-off to insertion into a parking orbit at 50,000 feet altitude are presented in figure 55. The transfer from this lunar orbit to insertion into the transearth trajectory is accomplished by the application of a velocity increment of 3110 ft/sec at the insertion point.

i. Transearth - The transearth trajectory is presented in figure 53.

j. Reentry - The return perigee altitude is 120,000 feet, the velocity at perigee is 36,320 ft/sec, and the reference reentry altitude is 400,000 feet. The time at which the reentry altitude is reached is 4:00 p.m. local time, for the primary flight plan of 167 hours after launch. A ground track of the transearth and reentry phase of the flight plan is shown in figure 53. The possible landing area extends from the Western Pacific across the Southern United States and into the South Atlantic. The characteristics of the Spacecraft during reentry are for an L/D ratio of .5 and a W/C_DA of 50. The characteristics of the flight plan during reentry are presented in figure 56.

3.4 SPACECRAFT SYSTEMS.- A description of the characteristics of the Spacecraft and its systems is presented in this section.

3.4.1 Spacecraft Configuration.- The physical relationship of Spacecraft Modules and major components is specified graphically by schematics with identifying notes. Precise arrangements and detailed mechanical features are not intended to be inferred by the figures.

3.4.1.1. General Arrangement.- The Spacecraft arrangement for lunar landing missions is shown in figure 57.

3.4.1.2 Mission Arrangements.- Spacecraft arrangements for the various missions up through lunar landing are shown in figures 58, 59, 60, 61. These arrangements demonstrate system buildup and off-loading techniques convenient to component development and Launch Vehicle capabilities.

3.4.1.3 Command Module.- The Command Module physical features are defined by aerodynamic and heating performance requirements and crew utility and well-being considerations.

3.4.1.3.1 Geometric Characteristics.- The basic external geometry of the Command Module is shown in figure 62. The Command Module shall be a symmetrical, blunt body developing a hypersonic L/D of approximately 0.50. The L/D vector shall be effectively modulated in hypersonic flight. The modulation is achieved through constant c.g. offset and roll control.

3.4.1.3.2 Inboard Profile.- Basic arrangements of internal features fundamental to full utilization of the Command Module geometry are shown in figures 63, 64, 65, 66.

3.4.1.3.2.1 Load Mitigation Swept Volume.- The crew is suspended on discrete load mitigation devices which normally act on earth-landing impact. The swept volume displayed by this load mitigation stroke is significant and is to be recognized in the internal layout.

3.4.1.3.2.2 Crew Space Equipment.- Crew space equipment shall be free of protrusions and snags.

3.4.1.3.2.3 Center-of-Gravity Management.- Consideration shall be given to the arrangement of water stores to permit center-of-gravity management. Alteration of crew positions may be used for center-of-gravity management where orientation with respect to displays and controls is not limited.

3.4.1.3.2.4 Center Aisle.- The center crew support equipment is readily removable and stowable. This provides a center aisle which is required to make full use of the volume and give ready access to all regions.

3.4.1.3.2.5 Air Lock Operation.- An air lock is extended into the center aisle region for transient conditions and allows exit or egress to or from the Command Module in the environment of space from either side. See figure 63.

3.4.1.3.2.6 Head Room.- Ground and flight crews performing maintenance, repair and checkout tasks have good head room resulting in an efficient operation. See figure 63.

3.4.1.3.2.7 Stations.- A variety of crew station combinations are obtained using various arrangements of individual stations. See figure 63.

3.4.1.3.2.7.1 Left Hand Station.- The left hand station is to be semi-permanently fixed in the near launch condition. Capabilities for movements to better utilize displays and to achieve comfort are to be provided. Access to equipment on the outboard side is achieved by a movement capability which uncovers the area of concern.

3.4.1.3.2.7.2 Center Station.- The center station is stowable and may be replaced by many combinations of crew orientation during flight.

3.4.1.3.2.7.3 Right Hand Station.- The right hand station can have the same capabilities as the left hand station or it can be stowed as is the center station depending on mission requirements.

3.4.1.3.2.8 Visibility.- A broad field of view is provided by windows over the crew's heads in the launch condition. These windows are covered by heat protection elements during launch, reentry and general flare activity at the discretion of the crew.

3.4.1.3.2.9 <u>Access and Egress Hatches</u>.- Three outward opening hatches are provided in the Command Module just above the crew's heads. The windows for broad field of view are incorporated in these hatches. All hatches are used for ground access, servicing and maintenance. The three hatches provide the crew with individual bailout or other types of emergency egress without interfering with each others activity. Normal access and egress for crew and all onboard equipment installation is achieved through the use of the large center hatch.

3.4.1.3.2.10 <u>Landing System</u>.- The Landing System is stowed in the upper portion of the afterbody. See figure 66.

3.4.1.3.2.11 <u>Reaction Control</u>.- The tankages and other impact sensitive elements for the Reaction Control System are stowed out of the nominal impact area. See figure 66.

3.4.1.3.2.12 <u>Load Mitigation</u>.- The nominal impact area is provided with load mitigation structure which absorbs the initial energy of impact for the Command Module. See figure 66.

3.4.1.4 <u>Spacecraft Adapter</u>.- The method of attachment, basic structure and external geometry with the exception of length shall be identical between Spacecraft Adapters for all mission configurations. The adapter structure shall be compatible with the adjoining modules and propulsion stages. Its overall bending stiffness shall satisfy the requirements of the Launch Vehicle. Its construction shall be buffet and noise resistent in atmospheric phase of flight.

3.4.1.5 <u>Space Laboratory Module</u>.- Spacecraft arrangements utilizing a Space Laboratory Module are limited to earth-orbital missions. A basic geometry of the Space Laboratory Module is shown in figure 59. External geometry of the Space Vehicle containing the Space Laboratory Module is held uniform with other nonlunar landing mission configurations. An arrangement for "docking" the Command Module and Space Laboratory Module is shown in figure 67.

3.4.2 Command and Service Modules Systems. - The characteristics of the major systems included in the Command and Service Modules are presented.

3.4.2.1 Guidance and Control System. - The Guidance and Control System is comprised of a Guidance and Navigation System and a Stabilization and Control System. The Stabilization and Control System provides the attitude stabilization and maneuver control requirements for the Spacecraft and for combinations of Spacecraft and appropriate Propulsion Modules. The Guidance and Navigation System provides steering and thrust control signals for the Stabilization and Control System, Reaction Control Systems, and appropriate propulsion system and their respective gimbal systems.

3.4.2.1.1 Navigation and Guidance System. -

3.4.2.1.1.1 Functional Requirements. - The functional requirements of the Navigation and Guidance System are presented below.

3.4.2.1.1.1.1 Space Vehicle Guidance. - The Navigation and Guidance System shall be capable of controlling the injection of the Spacecraft and of providing a monitoring capability of injection guidance to the crew. This shall be accomplished for both direct ascent and for injection from a parking orbit.

3.4.2.1.1.1.2 Midcourse Guidance. - The Navigation and Guidance System shall provide navigation data and compute velocity corrections in ciscular space to achieve terminal conditions at the moon and earth which allow a safe lunar landing and earth reentry, respectively. Enroute to the moon a mission abort capability shall be provided.

3.4.2.1.1.1.3 Reentry Guidance. - The Navigation and Guidance System shall be capable of guiding the Command Module during reentry through the earth's atmosphere to a preselected landing site on the earth. This capability shall be provided for reentry from lunar missions and earth orbits, from preinjection aborts, and from postinjection aborts.

3.4.2.1.1.1.4 Lunar Orbit and Lunar Landing. - The Navigation and Guidance System shall provide a capability for establishing lunar orbits and making lunar landings from orbit. An abort capability shall be provided from the lunar maneuver.

3.4.2.1.1.1.5 Lunar Takeoff. - The Navigation and Guidance System shall provide the capability of launch from the surface of the moon into an earth return trajectory by both direct ascent and by a parking orbit.

3.4.2.1.1.1.6 Rendezvous. - The Navigation and Guidance System shall be capable of accomplishing a rendezvous in earth orbit between the spacecraft and the Space Laboratory or other cooperative Space Vehicles.

3.4.2.1.1.2 System Description. - The system shall achieve simplicity and reliability by effectively employing the crew whenever equipment design advantage and crew capability are compatible. The system shall achieve operational versatility but, when versatility results in disproportionate increase in equipment complexity, onboard versatility shall be sacrificed and reliance shall be placed upon ground assistance. The system shall be reliable but reliability shall be obtained by the use of system or subsystem redundancy only if it cannot be obtained by ground cooperation and/or onboard emergency systems.

3.4.2.1.1.3 Subsystems. - Subsystems which have been clearly identified are the following:

Inertial platform
Space sextant
Computer
Controls and displays
Electronics assembly
Charter and star catalog

In addition to the above, there are requirements for range and/or velocity measuring equipment for terminal control in rendezvous and lunar landing. These may require either radar or visual range-finding equipment or both. There are also requirements for backup inertial components for emergency operation.

3.4.2.1.2 Stabilization and Control System. - The functional requirements and a description of the Stabilization and Control System are presented below.

3.4.2.1.2.1 Requirements. - The system shall satisfy the following requirements.

3.4.2.1.2.1.1 Atmospheric Abort. - Flight-path control during the thrusting period of atmospheric abort and stability augmentation after Launch Escape Propulsion System separation.

3.4.2.1.2.1.2 Extra-Atmospheric Abort. - Orientation, attitude control, and reentry stabilization and control. The system shall accept commands from the guidance system for thrust vector control and reentry control.

3.4.2.1.2.1.3 Parking Orbit. - Stabilization of the Spacecraft plus the lunar injection configuration while in a parking orbit.

3.4.2.1.2.1.4 Translunar and Transearth. - Stabilization and control during midcourse flight both outboard and inboard. The control technique shall provide fuel economy and shall satisfy all navigation requirements as well as solar orientation and antenna-pointing requirements. Attitude control and orientation for application of midcourse corrections shall be provided.

3.4.2.1.2.1.5 Orbital Rendezvous and Docking. - Rendezvous and "docking" with the Space Laboratory or other cooperative space vehicles.

3.4.2.1.2.1.6 Lunar Landing and Take-Off. - Attitude control and hovering for accomplishing landings and take-offs from the moon and for entering and departing from lunar orbits. Attitude control commands shall be accepted from the Navigation and Guidance System and the crew. Primary hovering control commands will be initiated by the crew.

3.4.2.1.2.1.7 Reentry. - Control requirements for reentry guidance. Reaction jets will be employed for three-axis stabilization. Reentry control will be provided by rolling the vehicle which is trimmed at an L/D.

3.4.2.1.2.1.8 Landing. - Stabilizing and controlling the Command Module with respect to the flight direction in the landing configuration. Command control will be by the crew employing visual reference.

3.4.2.1.2.1.9 Special Control Features. - Consideration shall be given to meeting a requirement for fine control of Spacecraft rolling response to tracking control commands from the Guidance and Navigation Systems. Consideration shall also be given to methods for optimizing overall system design for midcourse flight by integrating requirements for Spacecraft three-axis control and antenna-pointing requirements.

3.4.2.1.2.2 System Description.- The Stabilization and Control System shall consist of the following basic components:

Attitude reference
Rate sensors
Control electronics assembly
Manual controls
Attitude and rate displays
Power supplies

3.4.2.1.2.2.1 Attitude Reference.- The attitude reference system provides angular displacement signals to the Stabilization and Control System and instrument panel displays. The primary reference system is provided within the Guidance and Navigation System. Requirements for additional special attitude sensors are not specified at this time. Division of responsibility between Principal Contractor and Guidance and Navigation Contractor will be determined by NASA after sufficient study and interface negotiations. Some examples of these requirements follow.

3.4.2.1.2.2.1.1 Standby Inertial Reference.- A standby reference which is capable of retaining an inertial reference throughout any combination of Spacecraft maneuver. This system may be erected by the primary reference system but it must be capable of having erected to an inertial reference independent of the primary navigation system. It should also be capable of driving panel displays.

3.4.2.1.2.2.1.2 Special Sensors.- Non-gyroscopic sensors are required for solar orientation during midcourse flight and for third-axis control in connection with antenna-pointing requirements. Consideration should be given to the use of the outputs of these sensors to control directly through the switching logic of the electronic assembly and to the use of derived rate from the sensor output.

3.4.2.1.2.2.2 Rate Sensors.- Three axes rate gyro packages shall provide stability augmentation during propulsion modes, maneuvers and reentry. They also serve as a necessary sensor for the Rate Command System and require a dynamic range capable of dealing with all vehicle configurations and mission requirements. Redundancy shall be provided compatible with the overall system configuration.

3.4.2.1.2.2.3 Control Electronics Assembly.- The Stabilization and Control System Control Electronics Assembly shall accept command inputs from the Navigation and Guidance System during perio[redacted]ector control, periods of tracking for navigation purposes, and from the Stabilization and Control System attitude reference at all other times. The Control Electronics Assembly shall supply thrust command signals to the attitude control propulsion motors to establish correct orientation, stable limit cycle operation, and damping throughout all phases of the mission. The control electronics shall use pulse modulation or similar techniques by which the desired objectives of economical limit cycling, accurate control during periods of large disturbances, and satisfactory maneuver rates can all be achieved with the same switching logic. Flexibility to deal with all vehicle configurations and mission requirements shall be attained by the provision of adjustments for parameters such as attitude dead band, rate limits, and attitude to rate gain. To ease the control task of the pilot, the system must be capable of accepting discrete "dialled" orientation commands or provide an attitude followup for reengagement of attitude hold when the maneuver is completed. The control electronics shall be of modular construction and provide the necessary redundancy and inflight maintenance capability.

3.4.2.1.2.2.4 Manual Control.- The suggested method of maneuvering by the pilot is by opening the outer loop and imposing rate commands on the inner rate stabilization loop. The manual controls shall be capable of operating all reaction control motors by direct electrical connection, providing emergency operation in case of rate gyro or other automatic system failure. Design of the controls shall provide acceptable feel characteristics for all conditions of flight environment. Provision for the translational control by the crew for docking and hovering phases of the mission shall be compatible with the attitude control system concept.

3.4.2.1.2.2.5 Attitude and Rate Displays.- Angular rates and vehicle attitudes with respect to the reference shall be displayed during manual maneuvering and critical phases of the mission. At all other times, displays shall be commensurate with crew requirements.

3.4.2.1.2.2.6 Power Supplies.- The Stabilization and Control System shall generate all levels of DC and AC voltage requirements internally from the basic vehicle electrical supply. Choice of operating frequencies and provision of redundancy in the power supplies shall be governed by the requirements for compatibility between Navigation and Guidance and Stabilization and Control Systems.

3.4.2.2 Service Propulsion System. -

3.4.2.2.1 General Description. - The Apollo Service Propulsion System will be located in the Service Module and be capable of meeting the requirements for:

a. Abort propulsion after jettison of Launch Escape Propulsion System.

b. All major velocity increments and midcourse velocity corrections for missions prior to lunar landing mission.

c. Lunar launch propulsion and transearth midcourse velocity correction for return from lunar landing mission.

A suggested single engine schematic configuration is shown in figure 68. The Service Propulsion System will utilize earth-storable, hypergolic propellants, will include single or multiple thrust chambers with a thrust-to-weight ratio of at least 0.4 for all chambers operating, (based on the lunar launch configuration), and will have a pressurized propellant feed system.

3.4.2.2.2 Performance Requirements. - The Service Propulsion System will be required to function in several normal and emergency modes, depending upon the mission for which it is being used.

3.4.2.2.2.1 Earth Orbit Mission. -

3.4.2.2.2.1.1 Earth-Orbital Retrograde Velocity. - A retrograde velocity is required to reenter the Spacecraft from earth-orbital mode.

3.4.2.2.2.1.2 Rendezvous Velocity. - Depending upon the mission description, a velocity increments is required for rendezvous maneuvers in earth orbit.

3.4.2.2.2.1.3 Earth Orbital Corrections. - A velocity increment is required to correct earth-orbit after insertion by Launch Vehicle.

3.4.2.2.2.1.4 Post-Atmospheric Abort. - Mission abort during post-atmospheric portion of launch trajectory will be accomplished by the Service Propulsion System. As the launch escape propulsion system is to be jettisoned shortly after escape from the atmosphere, the Service Propulsion System will be utilized for velocity increment as required to provide separation from the Launch Vehicle, reentry control, and landing point selection if required.

3.4.2.2.2.2 Circumlunar Mission. -

3.4.2.2.2.2.1 Midcourse Velocity Corrections. - The mission trajectory selected will influence the magnitude of midcourse velocity corrections required for this mission. The Service Propulsion System will supply gross velocity increments not supplied by the Reaction Control System.

3.4.2.2.2.2.2 Rendezvous and Escape Requirements. - A velocity increments is required incidental to rendezvous with a launch vehicle stage in earth orbit and/or to augment this stage in attainment of escape velocity.

3.4.2.2.2.2.3 Launch Abort at Sub-Orbital Velocities. - Mission abort during post-atmospheric portion of launch trajectory will be accomplished by the Service Propulsion System. After the Launch Escape Propulsion System is jettisoned, the Service Propulsion System will be utilized for velocity increment as required to provide separation from the Launch Vehicle, reentry control, and landing point selection if required.

3.4.2.2.2.2.4 Launch Abort at Super-Orbital Velocities. - A velocity increments is required to separate the Spacecraft from the Launch Vehicle and to reorient the total velocity vector such as to allow early reentry or safe orbital stabilization.

3.4.2.2.2.3 Lunar Orbit Missions. -

3.4.2.2.2.3.1 Midcourse Velocity Corrections. - The mission trajectory selected will influence the magnitude of midcourse velocity corrections required for this mission. The Service Propulsion System will supply gross velocity increments not supplied by the Reaction Control System.

3.4.2.2.2.3.2 Lunar Capture. - In the vicinity of the moon the Service Propulsion System will supply velocity increment as required for insertion into lunar orbit from a free-return, circumlunar trajectory 5° out of the plane of the lunar orbit.

3.4.2.2.2.3.3 Lunar Orbit Transfer. - The velocity increment is as required to transfer from a circular to elliptical lunar orbit.

3.4.2.2.2.3.4 Lunar Escape Velocity. - The velocity increment is as required for lunar-orbit escape and earth return.

3.4.2.2.2.3.5 Post-Atmospheric Abort. - Mission abort during post-atmospheric portion of launch trajectory will be accomplished by the Service Propulsion System. After the Launch Escape Propulsion System is jettisoned the Service Propulsion System will be utilized for velocity increment as required to provide separation from the Launch Vehicle, reentry control, and landing point selection if required.

3.4.2.2.2.3.6 Super-Orbital Abort. - The velocity increment is as required to separate the Spacecraft from the Launch Vehicle and to reorient the total velocity vector such as to allow early reentry or safe orbital stabilization.

3.4.2.2.2.3.7 Post-Injection Abort. - The velocity increment is as required to separate the Spacecraft from the Launch Vehicle and to reorient the total velocity vector such as to allow early reentry or safe orbital stabilization.

3.4.2.2.2.4 Lunar Landing Mission. -

3.4.2.2.2.4.1 Lunar Launch. - The prime factor in the design of the Service Propulsion System is that it be capable of launching the Spacecraft from the lunar surface. It shall provide the velocity increment as required for lunar launch into an elliptical lunar orbit including the effects of the lunar gravitational field.

3.4.2.2.2.4.2 Lunar Orbit Transfer. - The velocity increment is as required to circularize an elliptical lunar orbit.

3.4.2.2.2.4.3 Lunar Escape Velocity. - The velocity increment is as required for lunar-orbit escape and earth return.

3.4.2.2.2.4.4 Post-Atmospheric Abort. - Mission abort during post-atmospheric portion of launch trajectory will be accomplished by the Service Propulsion System. After the Launch Escape Propulsion System is jettisoned, the Service Propulsion System will be utilized for velocity increment as required to provide separation from the launch vehicle, reentry control, and landing point selection as required.

3.4.2.2.2.4.5 Super-Orbital Abort. - The velocity increment is as required to separate the Spacecraft from the Launch Vehicle and to reorient the total velocity vector such as to allow early reentry or safe orbital stabilization.

3.4.2.2.2.4.6 Post-Injection Abort. - The velocity increment is as required to separate the Spacecraft from the launch vehicle and to reorient the total velocity vector such as to allow early reentry or safe orbital stabilization.

3.4.2.2.2.4.7 Lunar-Landing Abort. - At any time during the lunar capture or lunar-landing maneuvers, the Service Propulsion System must provide separation from the lunar-landing stage and safe return to earth.

3.4.2.2.3 System Operation. -

3.4.2.2.3.1 Operating Features.- Helium tanks shall be positively sealed by redundant valves prior to use to prevent leakage. Primary and secondary propellant pressurization regulators are required. Propellant utilization control is required to maintain proper oxidizer/fuel ratio. This may be accomplished through regulating either oxidizer or fuel flow rates. It is desirable to minimize the number of valves required to open to obtain ignition. Propellant utilization control should provide for primary and secondary mode of operation. If multiple oxidizer and fuel tanks are used, cross ties to allow use of primary or secondary controls with any tank of the regulated fluid (oxidizer or fuel) shall be provided. Isolation valves shall be provided in these tie lines. If multiple fuel and oxidizer tanks are used, each tank should be emptied prior to use of next tank. Automatic switch over from tank to tank with manual override is required. Propellant for large velocity requirements such as lunar launch and lunar escape shall be fed from the main propellant tanks. Propellant for small velocity requirements such as mid-course corrections will be fed from positive expulsion tanks common with the Service Reaction Control System or auxiliary positive expulsion tanks.

3.4.2.2.3.2 Safety Features. - Filters to protect regulators, control valves, and propellant injectors are required. Check valves to prevent oxidizer-fuel cross flow shall be provided in pressurization lines. There shall be relief valves to relieve high propellant tank pressures. Burst discs shall protect relief valves from propellant contamination. Redundant propellant valves are required. Manual override provisions on automatic solenoid valves and control valves shall be required. Helium and propellant tanks are to be sealed prior to use. Maximum use of welded or brazed lines and fittings to minimize leak points is desirable. Modular concept of replacement of subsystems is desirable.

3.4.2.2.3.3 Preflight Checkout. - Fittings are to be provided to gas leak check and purge all parts of the system and to flow check all regulators and control valves and to check normal and emergency valve sequence. Provisions for checking gimballing system prior to operation shall be required. Provisions for checks of all sensors and instrumentation required. Means for cleaning or replacement of system filters prior to flight shall be provided. System shall be static-fired as an entire system prior to flight, preferably in the lunar launch configuration including Command Module and other Service Module systems.

3.4.2.2.4 Crew Participation (Displays and Control). -

3.4.2.2.4.1 Monitoring Function. - During flight and non-operating periods the crew shall monitor tank pressures, leakage rates, and temperatures. Instrumentation of these items is therefore required. During engine operation, crew shall be able to monitor such system operating parameters as propellant and helium pressures, chamber pressure, and temperatures. Prior to and during engine operation, the crew shall be able to monitor engine gimbal operation. Crew shall be able to monitor all engine valving position. Remaining quantities of oxidizer and fuel shall be presented to crew during operation. Some means of indicating reserve propellant remaining and propellant utilization system operation are desired.

3.4.2.2.4.2 Normal Operation. - Preparation of system for engine ignition shall be a crew function. Actual firing shall be an automatic function performed by the guidance system. Crew shall be able to monitor the automatic function. Crew shall be able to monitor automatic switchover function from one oxidizer or fuel tank to another provided multiple tanks are used.

3.4.2.2.4.3 Emergency Operation. - There shall be a means provided to automatically and/or manually initiate system operation for abort capability in event of booster failure. Automatic engine shutdown of redundant engines in event of low chamber pressure, high external temperatures indicating external fire or other engine failures shall be provided. Automatic switchover to redundant regulator in event of helium primary regulator failure in the open mode is required to prevent helium loss. If multiple oxidizer and fuel tanks are used, manual override of automatic tank switching device shall be provided. Crew shall be able to manually shutdown the engine or engines and shall be able to manually override any automatic function. Provisions for in flight, extra-vehicular access by crew for inspection, and maintenance shall be provided.

3.4.2.2.5 Guidance System Requirements. -

3.4.2.2.5.1 Thrust-Vector Control. - This system shall be designed to control the thrust-vector during midcourse velocity corrections with frequent restart capability. Also, it shall control the thrust-vector during lunar launch. The gimbal actuators shall be redundant for this engine. These actuators shall have the necessary response characteristics to maintain proper trajectories.

3.4.2.2.5.2 Velocity Cut-off Control. - This system shall be designed to control the velocity-cutoff during midcourse velocity corrections. For lunar launch this system shall be designed and developed with an elapsed time from start signal to 90% of thrust of less than 300 milliseconds. The valve closing times should be held within limits to prevent damage from hydraulic hammer effect. Engine surge characteristics shall be investigated for different valve designs. Engine cutoff impulse accuracy shall be known within 2% of that required for the minimum operating cycle. The magnitude of impulse error shall not exceed 150 lb-sec.

3.4.2.2.6 Propellants. - The Service Propulsion System shall use an earth storable hypergolic bipropellant combination.

3.4.2.2.6.1 Oxidizer. - The oxidizer shall be nitrogen tetroxide (N_2O_4) with nitrous oxide (N_2O) added to depress the freezing point if necessary.

3.4.2.2.6.2 Fuel. - The fuel shall be either monomethylhydrazine (MMH) or a mixture of 50% hydrazine (N_2H_4) and 50% unsymmetrical dimethylhydrazine (UDMH).

3.4.2.2.7 Component Selection. - The example schematic shows a single thrust chamber. The schematic indicates the essential components and their functional interelationship.

3.4.2.2.7.1 Tanks. - Toroidal tanks shall not be used with multiple thrust chambers.

3.4.2.2.7.2 Pressurization. - Ambiently stored helium gas shall be used for pressurization. Tanks shall be held to minimum consonant with packaging requirements.

3.4.2.2.7.3 Control Systems. -

3.4.2.2.7.3.1 Thrust Vector Control. - Thrust vector control shall be attained by redundant gimbaling. Electromechanical and electrohydraulic systems should be compared.

3.4.2.2.7.3.2 Liquid Level Sensing System. - A liquid level sensing system is required to indicate propellant level in the tanks during periods of acceleration so that the crew can check propellants available.

3.4.2.2.7.3.3 Propellant Utilization System. - A propellant utilization system shall be supplied to ensure efficient utilization of propellant.

3.4.2.2.7.4 Performance. - The minimum delivered vacuum specific impulse shall be 315 $\frac{\text{lb-sec}}{\text{lb}}$.

3.4.2.3 Reaction Control System.- The Command and Service Modules shall include Reaction Control Systems to provide the impulse for attitude control and stabilization. The Service Module System shall also be capable of minor translational velocity increments.

3.4.2.3.1 Command Module Reaction Control System.- This system will be used only after separation of the Command Module from the Service Module.

3.4.2.3.1.1 Requirements.- The system shall provide three-axis control prior to the development of aerodynamic moments, roll control during reentry and landing, and pitch and yaw rate damping during reentry and deployment of the landing system. A roll acceleration of at least 10°/sec/sec shall be provided during reentry. The pitch and yaw acceleration shall be compatible with their requirements.

3.4.2.3.1.2 Description.-

3.4.2.3.1.2.1 General.- The suggested Reaction Control System is pulse modulated, pressure fed, and utilizes earth storable hypergolic fuel. Fuel tanks shall be positive-expulsion type. The Command Module has two independent systems as shown in figure 69 and located as shown in figure 70. Each is capable of meeting the total torque and propellant storage requirements. Each system consists of helium pressurization, propellant storage, distribution and thrust chamber subsystems.

3.4.2.3.1.2.2 Propellants.- The Reaction Control System shall use an earth storable hypergolic bipropellant combination.

3.4.2.3.1.2.2.1 Oxidizer.- The oxidizer shall be nitrogen tetroxide (N_2O_4) with nitrous oxide (N_2O) added to depress the freezing point if necessary.

3.4.2.3.1.2.2.2 Fuel.- The fuel shall be either monomethylhydrazine (MMH) or a mixture of 50% hydrazine (N_2H_4) and 50% unsymmetrical dimethylhydrazine (UDMH).

3.4.2.3.1.2.3 Distribution.- Each system shall consist of 6 thrust chamber subsystems installed to provide 6 degrees of control as shown in figure 69. Either system shall be capable of remote isolation by the crew by means of reversible valves.

3.4.2.3.1.2.4 Thrust Chambers. - Since the chambers and nozzles must be buried within the module, they should be ablatively cooled. Alternate methods may be considered which would also maintain compatible external temperatures on the units. Heat protection of control valves shall be provided if necessary.

3.4.2.3.1.3 Operation. -

3.4.2.3.1.3.1 Servicing and Checkout. - Each system shall be designed to allow preflight servicing, checkout, and deactivation.

3.4.2.3.1.3.2 Flight. - The systems shall be designed to operate simultaneously from the crew manual controls by means of electrical outputs. Malfunction detection means should be investigated which would allow for shutdown of one system if a thruster in that system fails.

3.4.2.3.2 Service Module Reaction Control System. - This system will provide the impulse for attitude control and stabilization for the Space Vehicle in all phases of flight except during periods that other propulsion systems are active. In addition, the system shall provide attitude control and stabilization for the Launch Vehicle Spacecraft combination in earth parking orbit. The system shall also provide minor translational capability for minor midcourse corrections, terminal rendezvous and docking as well as ullage accelerations for Mission Propulsion System, if necessary. The system shall contain the flexibility required to allow its use in all missions.

3.4.2.3.2.1 Requirements. - The final requirements for the system will be coordinated with the Contractor as they become available. A preliminary estimate indicates that 400 pounds of propellant including 30 percent reserve will be adequate for all missions, provided a minimum pulse not to exceed 20 milliseconds with a 90 percent pulse efficiency is attained.

3.4.2.3.2.2 Description. -

3.4.2.3.2.2.1 General. - The Reaction Control System is pulse modulated, pressure fed, and utilizes earth-storable hypergolic fuel. Fuel tanks shall be positive-expulsion type. The suggested Service Module Reaction Control system has two independent systems as shown in figure 71 and located as shown in figure 72. Each is capable of meeting the total torque and propellant storage requirements. The system shall consist of 8 roll, 4 pitch and 4 yaw, thrust chamber subsystems. It shall be designed to allow translation in six directions. The system has two sets of propellant tanks and pressurization subsystems. The thrust shall be installed so that normal translational thrust vectors are in the vicinity of the vehicle center of gravity for the earth-orbiting mission. The roll thrusters will be used for pitch and yaw maneuvering whenever the Lunar Landing Module is attached as well as in earth-parking orbit with the booster upper stage attached. Each set of tanks supplies six small thrust chambers capable of control in all 6 directions during navigational sightings.

3.4.2.3.2.2.2 <u>Propellant</u>.- The propellants shall be the same as those used for the Service Propulsion System.

3.4.2.3.2.2.3 <u>Distribution</u>.- Each set of tanks normally supply one thruster of each couple and the feed systems from the two sets of tanks are normally isolated. Figure 71 shows a system utilizing positive expulsion tanks. The method of storage must satisfy Reaction Control and minor mission velocity requirements. Each set of thruster chambers is capable of being isolated by the crew.

3.4.2.3.2.2.4 <u>Thrust Chambers</u>.- An intensive program should be undertaken to develop and evaluate both the radiation-cooled and ablation-cooled thrust chambers for this application. Particular attention should be given to the effects of intermittent firing of the chambers in a hard vacuum.

3.4.2.3.2.3 <u>Operations</u>.-

3.4.2.3.2.3.1 Servicing and Checkout.- Each system shall be designed to allow preflight servicing, checkout, and de-activation.

3.4.2.3.2.3.2 Flight.- The system shall be designed to operate from manual-electric and automatic-electric input signals. A major effort is required in order to develop satisfactory means for malfunction detection in the system especially in the area of small leakage rates.

3.4.2.4 Launch Escape System.- The Command Module shall be fitted with a Launch Escape System as shown in figure 57.

3.4.2.4.1 Requirements.- The Launch Escape Propulsion System separates the Command Module from the Launch Vehicle in the event of failure or imminent failure of the Launch Vehicle during all atmospheric phases. The performance of the Launch Escape Propulsion System is dictated by the requirements of crew response and/or of the Abort Sensing Implementation System of the Launch Vehicle and the structural capability of the Command Module to resist overpressures due to Launch Vehicle explosion. Two critical flight modes are recognized.

3.4.2.4.1.1 Pad Escape.- For escape prior to or shortly after lift-off, the Launch Escape Propulsion System separates the Command Module from the Launch Vehicle and propels the Command Module to an altitude of at least 5000 feet and a lateral range at touchdown of at least 3000 feet without exceeding the crew tolerances. Stabilization and lateral control, if required, shall be provided.

3.4.2.4.1.2 Maximum Dynamic Pressure Escape.- For escape at maximum dynamic pressure, the Launch Escape Propulsion System separates the Command Module from the Launch Vehicle during thrusting of the Launch Vehicle and propels the Command Module a safe distance from the Launch Vehicle. The Launch Escape Propulsion System and the Command Module combination are aerodynamically stable or neutrally stable and have sufficient lateral control to obtain the maximum possible Launch Vehicle "miss" distance consonant with the crew tolerances.

3.4.2.4.2 Propulsion.- The basic propulsion system is a solid-fuel rocket motor with "step" or regressive burning characteristics. Its nozzles are canted to avoid direct impingement of the exhaust jets on the Command Module.

3.4.2.4.3 Stabilization and Control.- Stabilization and lateral control shall be provided.

3.4.2.4.4 Escape System Jettison.- The Launch Escape Propulsion System is jettisoned at approximately maximum altitude after "pad escape," or an appropriate time after high dynamic pressure escape, and is separated from the Command Module by a solid-fuel rocket motor. For normal flights, separation is effected by the main propulsion motor during early operation of the second stage of the Launch Vehicle.

3.4.2.4.5 Initiation and Control Mode Selection.- Initiation of escape and subsequent selection of control modes is the responsibility of the crew. There shall be no responsibility assigned to ground control or automatic systems unless there is insufficient time and/or information for crew action.

3.4.2.5 Earth Landing System.- The Command Module includes an Earth Landing System to be used under all flight conditions for earth landing requirements. It is also compatible with the use of a moderate L/D terminal landing system such as the "parawing," gliding parachutes, or rotors.

3.4.2.5.1 Requirements.- The system satisfies the following requirements after normal reentry, maximum dynamic pressure escape, and pad escape.

3.4.2.5.1.1 Postentry Stabilization.- Stabilizes the Command Module during postentry descent.

3.4.2.5.1.2 Velocity Control.- Reduces the vertical touchdown velocity to not more than 30 feet/second at an altitude of 5000 feet.

3.4.2.5.1.3 Impact Attenuation.- Reduces impact acceleration such that neither the Command Module primary structure or flotation is impaired. Any further attenuation required by the crew shall be provided by individual, crewman shock-attenuation devices.

3.4.2.5.1.3.1 Impact Attitude.- For nominal land landings the capsule should impact at an angle of -15° with the c.g. forward. See figure 66. This locates the crew in a feet first position. For water landings an impact of 15° with the c.g. aft and the crew located in head-first position is desirable. The maximum emergency limit "g" forces must not be exceeded for any landing regardless of capsule orientation.

3.4.2.5.1.4 Postlanding.- The system provides any necessary flotation, survival, and location aids.

3.4.2.5.2 Description.- The landing system consists of 2 FIST-type drogue chutes deployed by mortar and a cluster of three simultaneously deployed landing parachutes. Landing parachutes are sized such that satisfactory operation of any two of the three will satisfy the vertical velocity requirement. The Command Module is hung in a canted position from the parachute risers and oriented through a roll control to favor impact attenuation.

3.4.2.5.3 Initiation and Control.- Initiation of all functions can be manually controlled. Command Module roll orientation prior to impact can be also manually controlled.

3.4.2.6 Structural System.- In addition to the fundamental load carrying structures, the Command and Service Modules Structural System shall include meteoroid protection, radiation protection inherent in the structure, and Passive Heat Protection Systems. Primary structures shall be designed and evaluated in accordance with standard aircraft practice with the exception that no structure shall require pressure stabilization.

3.4.2.6.1 Command Module.-

3.4.2.6.1.1 Reentry Thermal Protection.- The Command Module's external thermal protection shall utilize the planned degradation of suitably reinforced plastics. The forebody shall utilize a charring ablator capable of sustaining high surface temperatures and providing effective blockage of external heat fluxes. Adequate reinforcement of the shield shall be provided to ensure shield integrity and satisfactory performance through all phases of flight.

Afterbody heat protection shall also utilize planned material degradation but consideration should be given to providing protection which is better suited to the more moderate heat flux environment and thinner gauges of the afterbody.

Passive control of heat fluxes to the interior of the Command Module shall be utilized. Circulatory heat exchange system used only for internal cabin comfort control.

Egress hatches, windows, umbilicals, etc., shall be located on low heat flux regions of the Command Module when possible and shall be covered by doors fairing smoothly to the capsule contour.

The reinforced plastic shall be mounted on a relatively rigid brazed or welded sandwich construction support capable of withstanding temperatures considerably in excess of conventional bonding temperatures. The adhesive bond between the ablator and the sandwich support must be capable of withstanding the low temperatures during the space flight and shall not be subjected to temperatures during reentry beyond demonstrated state-of-the-art capabilities.

3.4.2.6.1.2 Pressure Vessel.- The pressure cabin will be separate from the thermal protection system and will consist of an aluminum shell with longerons to react the concentrated loads from the escape tower and parachute attachments, and act as edge members for windows and hatches. Consideration will be given to the venting of the space between the pressure cabin and thermal protection. Viewing ports will form an integral part of the exit hatch.

3.4.2.6.2 Service Module.- The Service Module shall consist of a sandwich shell compatible with the noise and buffet requirements and the meteorite penetration requirements. In addition, it should maintain structural continuity with adjoining modules and be compatible with the overall bending stiffness requirements of the Launch Vehicle.

3.4.2.7 Crew Systems.-

3.4.2.7.1 Flight Crew.-

3.4.2.7.1.1 Size and Number.- The flight crew shall consist of three men. The size of each crew member shall be between the 10th and 90th percentile as defined in reference.

3.4.2.7.1.2 Division of Duties.- Tasks shall be so apportioned as to make maximum utilization of all three crew members. During launch, entry, and similar critical mission phases, division of key tasks amongst the three crew members shall be as nearly equal as possible. Division of specific responsibilities shall be as follows:

3.4.2.7.1.2.1 Commander (Pilot).- He shall control the vehicle in manual or automatic mode, during all phases of the mission. He shall select, implement and monitor the modes of navigation and guidance. He shall monitor and control key areas of all systems during time critical periods. He shall occupy either the left or center couch during launch and reentry.

3.4.2.7.1.2.2 Co-Pilot.- The co-pilot shall be second in command of the vehicle. He shall support the commander as alternate pilot and navigator. During critical mission phases, he shall monitor certain critical parameters of the spacecraft and propulsion systems. He shall occupy either the left or center couch during launch and reentry.

3.4.2.7.1.2.3 Systems Engineer.- During critical mission phases he shall monitor certain critical parameters of the spacecraft and propulsion systems. When certain systems are placed on board primarily to be evaluated for later Apollo vehicles, he shall be responsible for their operation, monitoring and evaluation. He shall occupy the right hand couch during launch and reentry.

3.4.2.7.1.2.4 General Duties.- All crew members shall be cross-trained so as to be able to assume the tasks usually performed by fellow crew members. Each shall stand watches during non-critical mission phases and perform all command and systems monitoring functions during such watches. While the commander shall be the principal navigator, the taking of navigational fixes and performance of associated calculations may be divided equally amongst all crew members.

3.4.2.7.2 Crew Integration.-

3.4.2.7.2.1 Displays and Controls.-

3.4.2.7.2.1.1 Arrangement.- Arrangement of displays and controls shall reflect the division of crew tasks. Critical piloting displays and controls shall be duplicated at the adjacent command and co-pilot stations. Non-duplicated piloting displays and controls shall be readily visible and accessible to both commander and co-pilot. All controls shall be of the self-locking type, or guarded to prevent inadvertent actuation. Instrument mountings shall ensure legibility of necessary displays during periods of vibration.

3.4.2.7.2.1.2 Manual Controls.- Manual control inputs to the Spacecraft attitude control system shall be provided at the left hand and center seats.

3.4.2.7.2.1.3 External View Devices.- Windows and other external viewing devices shall be provided to permit maximum feasible use of direct vision during rendezvous, lunar and earth landing, scientific observations and monitoring of crewmen operating outside the Spacecraft.

3.4.2.7.2.1.4 Operation by Single Crew Member.- Controls and displays shall be so arranged as to permit one crew member to return the vehicle safely to earth. However, this requirement shall not cause systems designs which result in degraded reliability for three man operation.

3.4.2.7.2.2 Crewspace Arrangement.-

3.4.2.7.2.2.1 Primary Duty Stations.- The primary displays, controls, and support systems shall be so arranged that the crew members are generally side-by-side during launch, entry, and similar critical mission phases. During other mission phases at least one couch shall be completely removed or stowed in order to provide additional work space and access to other work areas within the Command Module.

3.4.2.7.2.2.2 Secondary Duty Stations.- Areas for taking navigation fixes, performing maintenance, food preparation, and certain scientific observations may be separate from the primary duty stations.

3.4.2.7.2.2.3 Watch Station.- One of the primary duty stations shall be the station at which one crew member stands watch during non-critical mission phases. Some immediate control over all critical systems must be accessible at that station.

3.4.2.7.2.2.4 Sleeping.- There shall be a specific area assigned for sleeping. It shall accomodate a single crewman. It shall be placed in such a manner as to permit control of noise, light, and other distractions.

3.4.2.7.2.2.5 Toilet.- There shall be a specific area assigned to a toilet for collection of human waste. It shall accomodate a single crewman. It shall be so placed as to permit temporary partitioning for privacy.

3.4.2.7.2.2.6 Radiation Shielding.- The mass of the Spacecraft modules shall provide the majority of the bulk shielding. Arrangement of this mass shall be optimized to provide maximum shielding protection, both by its arrangement and by the position of the crew members **without unduly compromising the system.**

3.4.2.7.3 Crew Equipment.-

3.4.2.7.3.1 Acceleration Protection.-

3.4.2.7.3.1.1 Design Approach.- Design of the crew support and restraint systems shall be integrated with the design of the Earth-Landing and Launch Escape Propulsion Systems.

3.4.2.7.3.1.2 Couch.- Each crewman shall be provided with a support couch for protection against acceleration loads. The couch shall provide full body and head support during all nominal and emergency acceleration conditions. During launch and entry, the couch shall support the crew members at body angles specified in figure 73. The couch shall provide support during lunar landing, however, the crewmen may assume different positions at this time to facilitate control. The couch shall be adjustable to permit changes in body and leg angles to improve comfort during non-acceleration mission phases. Couches shall be so constructed as to permit crewmen to interchange positions. To meet the interchange requirements, sizing may be accomplished by use of simple couch adjustment devices. Couch construction and materials shall not amplify any accelerating forces by a factor of more than 1.2. The couch shall accomodate a crewman wearing a back type or seat type personal parachute. The parachute shall remain in the couch when the crewman leaves his restraint system. The couch shall accomodate crewmen in both pressurized and unpressurized pressure suits when used. The couch shall permit ease in ingress and egress during all nominal and emergency mission conditions. All couches shall be easily removable for the purpose of preflight and inflight maintenance.

3.4.2.7.3.1.3 Restraint System. - A Restraint System shall be provided with each couch. The system shall allow the interchange of crewman with simple attachment and adjustment for comfort and sizing. The torso portion of the Restraint System shall also serve as a personal parachute harness. The Restraint System shall provide adequate restraint for all nominal and emergency flight phases; landing loads and high dynamic pressure aborts are particularly significant in design of the Restraint System.

3.4.2.7.3.1.4 Impact Attentuation. - Impact attenuation beyond that required to maintain general Spacecraft integrity shall be obtained through use of discrete shock mitigation devices for individual crew support and restraint systems. Attenuation devices shall provide for lateral as well as transverse acceleration loads.

3.4.2.7.3.1.5 Vibration Attenuation. - Vibration attenuation beyond that required to maintain general Spacecraft integrity shall be provided with each support and restraint system. Such vibration attenuation systems must keep vibration loads transmitted to the crew within tolerance limits and also permit the crew to exercise necessary control and monitoring functions.

3.4.2.7.3.1.6 Restraint for Weightlessness. - An appropriate method of restraint shall be provided for a sleeping crew member.

3.4.2.7.3.2 Decompression Protection. - Pressure suits shall be provided for extra vehicular operations and in the event of cabin decompression. The same pressure suits shall be utilized for extra vehicular operations and cabin decompression emergencies. Mission reliability and crew safety requirements shall be satisfied without the use of pressure suits for cabin decompression. No beneficial effect on calculated reliability or crew safety shall be included in the analysis; nor shall there be any unrealistic compromise of Spacecraft systems imposed wholly by the use of pressure suits.

3.4.2.7.3.3 Sanitation. -

3.4.2.7.3.3.1 Human Waste. - The Spacecraft shall have a toilet for collection of urine and fecal waste. The toilet shall accomodate a single crewman. The toilet shall be placed in such a manner to permit use of temporary curtains or partitions for privacy. The collection system shall include means for disinfecting human waste sufficiently to render it harmless and unobjectional to the crew. All human waste shall be stored aboard the Spacecraft.

3.4.2.7.3.3.2 Personal Hygiene. - The Spacecraft shall be equipped with facilities for shaving, dental cleansing, bodily cleansing, and deodorizing. Facilities shall be included for cleansing of garments or for an appropriate number of garment changes.

3.4.2.7.3.3.3 Non-Human Waste. - The Spacecraft shall have provisions for handling of all other waste such as those from eating and personal hygiene.

3.4.2.7.3.3.4 Control of Infectious Germs. - The Spacecraft systems operation shall provide means for controlling infectious organisms which would have an unfavorable effect upon the crew members.

3.4.2.7.3.4 Food and Water. -

3.4.2.7.3.4.1 Food. - All food shall be of the dehydrated, freeze-dried or similar type that is reconstituted with water or does not require reconstitution. The food shall have a variety of flavor and texture similar to that provided in normal earth diets. There is no requirement for refrigerated storage; however, the foods shall require heating and chilling in preparation and service. The food items shall constitute a low bulk diet.

3.4.2.7.3.4.2 Water. - The primary source of potable water is from the fuel cell. In addition, sufficient water must be on board at launch to provide for the 72 hour landing requirement in event of early abort. Urine need not be recycled for potable water.

3.4.2.7.3.5 Emergency Equipment.-

3.4.2.7.3.5.1 Survival Equipment.- Post-landing survival equipment shall include one three-man life raft, food, location aids, first aid equipment and various accessories necessary to support the crew outside the Spacecraft for three days in any possible emergency landing area. Provisions shall be included for removing a three day water supply from the Spacecraft after landing; in addition, provisions shall be included for purifying a three day supply of sea water in event of water landing.

3.4.2.7.3.5.2 Personal Parachutes.- Each crewman shall be equipped with a personal parachute for use in event that the Spacecraft landing system malfunctions, cannot function, or cannot cope with local hazards. The personal parachute shall be stowed in the back or seat of each couch; the restraint harness shall serve as the parachute harness.

3.4.2.7.3.5.3 First Aid Equipment.- The Spacecraft shall be equipped with first aid and preventive medicine items for coping with various human injuries and disorders. If feasible, Spacecraft first aid equipment may be integrated with survival equipment first aid items.

3.4.2.7.3.6 Radiation Dosimeters.- Each crew member shall be provided with an accurate, simply read, personal dosimeter system. The dosimeter system shall be worn or placed immediately adjacent to the crew members at all times. Each system shall measure cumulative dose, shall contain a warning device, and shall have an output plug for telemetry signals.

3.4.2.7.3.7 Medical Instrumentation.-

3.4.2.7.3.7.1 Physiological Measurements.- Ultimate monitoring and telemetry requirements will be specified by NASA on the basis of early studies and operations. At this time it appears that the crew will perform all physiological monitoring of each other, except as noted below. Measurements shall be taken with simple, clinical devices; significant findings shall be reported by voice. One physiological parameter may be sensed automatically and telemetered periodically.

3.4.2.7.3.7.2 Early Orbital Missions.- During early Apollo orbital flights, a variety of biological instrumentation will be required to enhance crew safety and assess the crew's tolerance to long term weightlessness. During stressful periods of such early flights, the following data may be telemetered:

Electrocardiogram	2 channels
Blood pressure	Intermittently: shares ECG channels
Respiration rate and volume	1 channel
Body temperature	Commutated

During non-stress mission phases the above may be recorded intermittently on a programmed basic. Electroencephalography may also be required during non-stress phases. In addition to the above, various special physiological experiments will be performed in the Spacecraft and Laboratory Module as part of the NASA-furnished scientific payload.

3.4.2.7.3.7.3 Extra-Vehicular Operations.- During lunar exploration or manned extra-vehicular pressure suit operations, the following data shall be transmitted from the extra-vehicular suit and monitored within the spacecraft; Voice, one physiological function (respiration, heart-beat, or electroencephalogram), and one to four environmental parameters (pressure, temperature, carbon dioxide, oxygen). The Spacecraft displays shall permit switching between and identification of simultaneously operating extra-vehicular suits.

3.4.2.7.3.8 Other Crew Equipment.-

3.4.2.7.3.8.1 Garments.- In addition to pressure suits, the crew shall be provided with garments for wear during normal mission phases. These garments shall be comfortable, close fitting, and free of areas that would snag on Spacecraft equipment. The garment must be wearable under the pressure suit. Each crewman shall be provided with a light-weight cap to protect his head and eyes from injury as a result of collision with Spacecraft equipment. This cap may be separate from the pressure suit helmet.

3.4.2.7.3.8.2 Exercise.- Equipment shall be provided to permit the crew to exercise and maintain physical condition while in a weightless state.

3.4.2.7.3.8.3 Recreation.- The crew shall be provided with items for amusement and entertainment during off-duty periods.

3.4.2.8 Environmental Control System.- The Command and Service Modules shall include the Environmental Control System which provides a conditioned, "shirtsleeve" atmosphere for the crew; provisions for pressure suits in event of cabin decompression; thermal control of all Command and Service Module equipment where needed; and provisions for charging self-contained extra-vehicular pressure suit support systems ("back packs").

3.4.2.8.1 Requirements.- The system shall satisfy the following:

3.4.2.8.1.1 Metabolic.- The following conditions shall be provided by the Environmental Control System.

Total cabin pressure (O_2 and N_2 mixture) 7 ± 0.2 psia

Relative humidity 40 - 70%

Partial Pressure CO_2 - maximum 7.6 mm Hg

Temperature $75^{\circ} \pm 5^{\circ}$ F

3.4.2.8.1.2 Pressure Suit Operation.- The system shall provide for the use of individual pressure suits. In event of cabin decompression, the system shall provide a conditioned oxygen atmosphere at 3.5 psia to the suits. The system must be capable of maintaining a cabin oxygen partial pressure of at least 3.5 psia for 5 minutes following a **single one-half-inch diameter puncture in the pressure compartment in addition to the normal structural design criteria.**

3.4.2.8.1.3 Equipment Cooling.- The system shall provide thermal control for equipment. No critical equipment shall depend upon the cabin atmosphere for cooling or depressurization.

3.4.2.8.2 System Description.- Environmental control is accomplished with two air loops, a gas supply system and a thermal control system. (See Fig. 74.)

3.4.2.8.2.1 Air Loops.-

3.4.2.8.2.1.1 Regenerative Circuit Loop.- This loop supplies the conditioned atmosphere to the cabin and/or pressure suits.

3.4.2.8.2.1.1.1 Particulate Removal.- A debris trap shall be provided.

3.4.2.8.2.1.1.2 Noxious Gases. - Noxious gases shall be removed by activated charcoal and a catalytic burner with the latter provided with a regenerative heat exchanger. A gas analyzer shall be provided.

3.4.2.8.2.1.1.3 Carbon Dioxide. - Carbon dioxide shall be absorbed by lithium hydroxide. The system shall provide for two parallel isolated lithium hydroxide canisters. The size of canisters and method of cartridge replacement shall be optimized.

3.4.2.8.2.1.1.4 Atmospheric Circulation. - The loop shall be provided with three parallel isolated blowers, any one of which will circulate the required flow. One operating blower shall be capable of supplying the following requirements to all three pressure suits simultaneously:

Ventilation flow at 3.5 psia	12 CFM thru each suit
Maximum flow resistance of each suit	5" of water at 12 CFM, 3.5 psia

Each pressure suit connection shall have a bypass which will permit individual manual flow control.

3.4.2.8.2.1.1.5 Temperature Control. - A liquid coolant heat exchanger system shall be provided to cool the circulating air below the required dew point for condensate removal and humidity control. A regenerative heat exchanger shall be provided for the crew to control their inlet air temperature. A water evaporator shall also be provided for cooling of circulating air in event of loss of coolant.

3.4.2.8.2.1.1.6 Humidity Control. - The condensed water vapor shall be removed by either of two parallel isolated separators. Air-driven centrifugal water separators shall be developed for use. The development of a sponge type water separator shall be pursued until the desired use of the centrifugal separator is unquestioned.

3.4.2.8.2.1.2 Cabin Loop. - The loop serves to provide cabin ventilation and thermal control during all phases of the mission and postlanding ventilation.

3.4.2.8.2.1.2.1 Atmospheric Circulation. - The loop shall be provided with two fans, either of which is capable of circulating the required flow and shall be designed to operate as an efficient exhaust fan during the post-landing phase. Snorkels shall be provided for post-landing.

3.4.2.8.2.1.2.1 Temperature Control.- A liquid coolant heat exchanger shall be provided to control cabin air temperature. It shall be designed to minimize fan power during the post-landing phase.

3.4.2.8.2.2 Gas Supply System.- The primary gas supplies shall be stored as super critical cryogenics in the Service Module. Storage of these supplies is discussed in connection with the Electrical Power System.

3.4.2.8.2.2.1 Primary Gas Requirements.- The gas supply shall have a 50 percent excess capacity over that required for normal metabolic and leakage needs, plus two complete cabin repressurizations, and a minimum of 18 air lock operations.

Provisions shall be made for recharging portable life support systems ("back packs").

3.4.2.8.2.2.2 Reentry Oxygen.- The Command Module shall contain a supply of gaseous oxygen in a high pressure bottle which shall be sufficient for reentry. A completely redundant system shall also be provided.

3.4.2.8.2.3 Thermal Control.- The normal dissipation of the internal thermal load of the Spacecraft is accomplished by absorbing heat with a circulating coolant and rejecting this heat from a space radiator during certain mission modes. Other cooling systems will supplement or relieve the primary system.

3.4.2.8.2.3.1 Radiator.- The space radiator shall be integral with the skin on the Service Module. For redundancy, dual coolant loops using radiator panels shall be provided. The radiator shall be designed to adequately meet the deep space random orientation condition. In addition, the radiator design shall be compatible with the water management program.

3.4.2.8.2.3.2 Coolant Loop.- The liquid coolant rejection transport fluid shall circulate through the regenerative circuit loop heat exchanger, electrical equipment cold plates, water evaporater gas storage heat exchanger, and the space radiator. Alternate liquid coolant passages in the equipment cold plates shall be provided. The liquid coolant flow shall be provided. The liquid coolant flow shall be maintained at fixed rate by one of three hermetically sealed, constant speed pumps. The redundant reservoir accumulator shall allow for a complete recharging of the liquid coolant. Provisions shall be made for a gas check before recharging in the event of a rupture to allow for isolation of leakage zones and to reestablish system integrity.

3.4.2.8.2.3.3 Mission Modes.-

3.4.2.8.2.3.3.1 Prelaunch (PAD).- Before lift-off the space radiator shall be isolated and the total heat load dissipated by cooling the liquid coolant in the water evaporator with ground support freon.

3.4.2.8.2.3.3.2 Launch.- After lift-off and attainment of sufficient altitude, water will be substituted in the evaporator for cooling.

3.4.2.8.2.3.3.3 Orbit.- The water evaporated in the liquid coolant loop may be used to supplement the radiator in earth and lunar orbit.

3.4.2.8.2.3.3.4 Transit.- The radiator shall be capable of dissipating the total heat load in spacecraft orientation during transit.

3.4.2.8.2.3.3.5 Lunar Landing.- A Refrigeration System shall be used to provide low temperature liquid coolant to the coolant loop. The high temperature heat rejection of the refrigeration unit shall be accomplished by the space radiator.

3.4.2.8.2.3.3.6 Reentry.- During reentry the thermal load shall be cooled by water evaporation in the liquid coolant heat exchanger. In event of liquid coolant loss, the metabolic heat load shall be cooled by a water evaporator in the regenerative suit-circuit loop.

3.4.2.8.2.3.3.7 Extra-Vehicular Pressure Suit Operations.- The Environmental Control System shall not directly support pressure suits during extra-vehicular operations. During such operations, the suits shall be supported by portable life support systems ("back packs").

3.4.2.8.2.4 Controls.- Control of atmospheric pressures, humidity and temperature shall be automatic with provisions for manual surveillance and control.

3.4.2.8.3 Water Management.- Water shall be collected from the separator and the fuel cell and stored in positive expulsion tanks. Water in excess of crews metabolic needs, sanitary needs, and a sufficient supply for cooling the total internal heat load on the moon for 24 hours shall be used as needed for supplemental cooling. The water collected from the fuel cells shall be stored seperately and used as the primary source of potable water.

3.4.2.8.3.1 Water Requirements.- Water shall be provided at lift-off to satisfy the crews postlanding metabolic needs and provide for evaporative cooling during exit and reentry following an immediate abort. A water management program shall be encompassed in the design to provide water requirements for all other phases of the mission.

3.4.2.8.4 Safety Features.- All relief valves, snorkel valves, and other valves which connect the internal pressure vessel to the space environment shall have manual closures and/or overrides. Filters shall be provided to protect all regulators, control valves, gas analyzers, etc. Relief valves shall be provided to prevent overpressurization of low pressure components. Flow limiting devices shall be provided to prevent excessive use of gas supplies and subsequent depletion of such supplies.

3.4.2.8.5 Preflight Checkout.- Fittings with proper access shall be provided to perform pressure checks, component performance tests, etc. during preflight checkout. This requirement is to preclude the necessity of breaking system integrity for component tests. Provisions shall be made for testing and calibrating all environmental sensors.

[redacted]

3.4.2.9 Electrical Power System. -

3.4.2.9.1 System Description. -

3.4.2.9.1.1 Purpose. - The Electrical Power System shall supply, regulate, and distribute all electrical power required by the Spacecraft for the full duration of the mission, including the post-landing recovery period.

3.4.2.9.1.2 Major Components. - The Electrical Power System shall be comprised of the following major components.

a. Three (3) non-regenerative hydrogen-oxygen fuel-cell modules.

b. Mechanical accessories, including control components, reactant tankage, piping, radiators, condensors, hydrogen circulators and water extractors, isolation valves and such other devices as required.

c. Three (3) silver-zinc primary batteries, each having a nominal 28 volt output and a minimum capacity of 3000 watt-hours (per battery) when discharged at the 10 hour rate at 80°F.

d. An Electrical Power System display and control panel, sufficient to monitor the operation and status of the system and for distribution of generated power to electrical loads, as required.

3.4.2.9.1.3 Location and Weight. - The location of each of the above components within the spacecraft shall be as listed herein. Every effort shall be exercised to minimize equipment size and weight, commensurate with the established requirements and obtaining the highest practicable reliability.

Component	Location
Fuel-cell module and controls	Service Module
Tanks (empty), Radiators, Heat exchangers, Piping, Valves	Service Module
Total Reactants, plus reserves	Service Module
Silver-Zinc Batteries	Command Module
Electrical power distribution and controls	Command Module

[redacted]

3.4.2.9.1.4 Operating Modes.-

3.4.2.9.1.4.1 Normal Operation.-During all mission phases, from launch until reentry, the entire electrical power requirements of the spacecraft shall be supplied by the three fuel-cell modules operating in parallel. The primary storage batteries would be maintained fully charged under this condition of operation.

3.4.2.9.1.4.2 Emergency Operation.-In the event of failure to one of the fuel-cell modules the failed unit would be electrically and mechanically isolated from the system and the entire electrical load assumed by the two fuel-cell modules remaining in operation. The primary batteries would be maintained fully charged under this condition of operation.

In the event of failure of two of the fuel-cell modules, the failed units would be electrically and mechanically isolated from the system. Spacecraft electrical loads would be immediately reduced by the crew and manually programmed to hold within the generating capabilities of the remaining operable Fuel-Cell Module. The primary batteries would be recharged, if necessary, and maintained fully charged under this operating condition.

3.4.2.9.1.4.3 Reentry and Recovery.-At reentry, the Fuel-Cell Modules and accessories will be jettisoned. All subsequent electrical power requirements shall be provided by the primary storage batteries.

3.4.2.9.2 System Requirements.-

3.4.2.9.2.1 Fuel-Cell Module.-Each Fuel-Cell Module shall have the following performance characteristics.

3.4.2.9.2.1.1 Type.-Fuel-Cell Modules shall be of the low pressure intermediate temperature, Bacon-type, utilizing porous nickel, unactivated electrodes and aqueous potassium hydroxide as the electrolyte. Fuel cells shall be operated non-regeneratively, utilizing hydrogen and oxygen as the reactants.

3.4.2.9.2.1.2 Output Power.-Each Fuel-Cell Module shall have a nominal capacity of 1200 watts at an output voltage of 28 volts and a current density conservatively assigned such that 50% overloads can be continuously supplied.

3.4.2.9.2.1.3 Pressure and Temperature.-The nominal cell operating pressure and temperature shall be approximately 60 psia and 425°F to 500°F respectively.

3.4.2.9.2.1.4 Fuel Consumption.-Under normal conditions of operation, the specific fuel consumption shall not exceed 0.9 lb/Kw-Hr, total H_2 and O_2.

3.4.2.9.2.1.5 Water Generation.-The water generated by the Fuel-Cell Module shall be potable and shall be separated from the hydrogen and stored.

3.4.2.9.2.1.6 Start Up.-Self-sustaining reaction within the Fuel-Cell Module shall be initiated at a temperature of approximately 275°F. Integral heaters shall be provided to facilitate ground starting as well as during the mission. These heaters shall not be capable of heating units to excessive temperatures with the fuel-cell and its cooling system inoperative.

3.4.2.9.2.1.7 Fuel-Cell Modules.-A detection and control system shall be provided with each Fuel-Cell Module to prevent contamination of the collected water supply.

3.4.2.9.2.1.8 System Redundancy.-The degree of redundancy provided for mechanical and electrical accessory equipment such as radiator loops, control valves, piping circuits, voltage regulator, etc., shall, in general, be 100 percent.

3.4.2.9.2.2 Electrical Distribution.-

3.4.2.9.2.2.1 General.-The distribution portion of the electrical power system shall contain all necessary busses, wiring protective devices, switching and regulating equipment.

Except as specified herein, the electrical distribution system shall conform to the requirements of standard MIL-STD-704.

Selection of parts and materials, workmanship, fabrication and manufacturing processes shall be guided by the requirements of MIL-E-5400, except as required to meet the performance or design requirement specified herein.

3.4.2.9.2.2.2 System Voltage.-Electrical power shall be generated and distributed at 28 volts DC (nominal).

3.4.2.9.2.2.3 Regulation.-The voltage level shall be regulated to prevent variance of more than ±2 volts from the nominal voltage under all conditions of operation of the fuel cell system.

3.4.2.9.2.2.4 AC Ripple.-All DC busses in the system shall be maintained essentially free of AC ripple (as defined by paragraph 3.12 of MIL-STD-704) to within a limit of 250 millivolts peak to peak.

3.4.2.9.2.2.5 Protection.-Busses and electrical loads shall be selectively protected such that individual load faults will not cause an interruption of power on the bus to which the load is connected. Likewise, a fault on the nonessential bus shall not cause an interruption of power to the essential bus.

3.4.2.9.2.2.6 Load Grouping.-All electrical loads supplied by the distribution system shall be classified as Essential, Nonessential, Pyrotechnic, or Recovery. Essential loads are defined as those loads (except pyrotechnic circuits) which are mandatory for safe return of the spacecraft to earth from any point in the lunar mission. Such loads as are not mandatory for safe return of the spacecraft shall be grouped on the Nonessential bus and provision made for disconnecting these loads as a group under emergency conditions. All loads required during the post-landing recovery period shall be supplied by the Recovery bus and provision made for manually disconnecting this bus from the Essential bus following landing. Redundant busses shall be provided for pyrotechnic circuits, and used to supply only that type load.

3.4.2.9.2.2.7 Power Conversion.-Equipment which requires conversion of basic electrical power (28 volts DC) to power with other characteristics shall accept the basic power as defined herein for modification and use. Conversion or inversion devices required for this purpose shall be integral with the utilization system or utilization equipment, whenever practical.

3.4.2.9.2.2.8 External Power.-Provision shall be made to energize the distribution system from an external source (28 volts, 100 amps DC) through an umbilical connector and a blocking diode.

3.4.2.9.2.2.9 Electrical Distribution Panel.-The Distribution Panel shall be dead front and adequately enclosed or otherwise protected to minimize hazards to the crew and provide maximum mechanical protection for the electrical system and components. Switching and control shall be accomplished by manually operated circuit breakers or contactors in preference to electrically operated contactors, except where use of a remotely controlled device is necessary to reduce the length of large electrical conductors.

3.4.2.9.2.2.10 System Type.-The distribution system shall be a two-wire grounded system, i.e., wire and busses shall be employed as the return path for electrical currents, in lieu of using the spacecraft structure for this purpose. The system negative shall be grounded at one point only and shall not be interrupted by any control or switching device.

3.4.2.9.2.3 Reactant Tankage.-

3.4.2.9.2.3.1 General Requirements.-Sufficient tankage shall be provided to store all reactants required by the Fuel-Cell Modules and environmental controls for a 14-day mission. Reactants shall be stored supercritically at cryogenic temperatures and the tankage shall consist of two equal volume storage vessels for each reactant. The main oxygen and nitrogen storage shall supply both the Environmental Controls System and the fuel-cells.

3.4.2.9.2.3.2 Reserves.-The tankage volume shall include the fuel-cell fluid requirements plus 10% reserve and the environmental fluid requirements. **The hydrogen storage volume shall include the fuel-cell requirements plus 10% reserve.**

3.4.2.9.2.3.3 System Arrangement.-Adequate valves and controls shall be provided to isolate identical reactant tanks from each other, and from the environmental controls and

Fuel-Cell Modules. Valve arrangement shall allow flow from any reactant tank to any Fuel-Cell Module.

The schematic arrangement Figure 75 is intended only to convey to the vendor the requirements covered above, rather than the complete system arrangement.

3.4.2.10 Communication and Instrumentation System.-

3.4.2.10.1 General Design Requirements.- Equipment shall be constructed to facilitate maintenance by ground personnel and by the crew. Each system, together with the interconnecting cables, shall be as nearly self-contained as possible to simplify removal from the spacecraft. The equipment and system shall be capable of sustained undergraded operation with supply voltage variation of +15 percent to -20 percent of the nominal bus voltage. Flexibility for incorporation of future additions or modifications shall be stressed throughout the design and assembly of all components and systems. Toward this end, the following features shall be provided:

a. Spare conductors shall be included in each wire group to permit system revisions or additions without necessitating retrunking of wire runs or additional bulkhead penetrations.

b. Insofar as possible, all spare contacts or relays, switches, contactors, etc., shall be wired and brought to an accessible point for future use, if needed.

A patch and programing panel shall be provided which will permit the routing of signal inputs from sensors to any selected signal conditioner and from these to any desired commutator channel. Panel design shall provide the capability of "repatching" during a mission.

3.4.2.10.1.2 Circuit Quality Analysis Chart.- The Contractor shall provide a circuit quality analysis for each radiating electrical system. The Contractor shall provide information showing exactly how ranging, telemetry, voice, and television data modulate all transmitters with which they are used. This information shall include:

a. Description of modulation systems used.

b. Bit rates used in each data mode.

c. Bandwidth required on subcarrier oscillators and main carrier.

d. Frequencies of all subcarrier oscillators and type of data on each.

3.4.2.10.2 Test and Maintenance.- The equipment and associated documentation shall be engineered for comprehensive and logical fault tracing. It shall be possible to check the operability of all functions of the equipment after installation in the Spacecraft. Each subsystem shall contain sufficient monitor points which are readily accessible to allow rapid and complete systems check. The equipment and systems shall be designed to facilitate prelaunch tests, before and after mating with the launch vehicle. Insofar as possible, the design shall provide for power control and system activation such that the maximum number of individual systems tests can be performed without full support or coordination with other spacecraft systems or those of the launch vehicle. It is of prime importance that the coupling of test equipment does not affect the on-board systems so that unrealistic test conditions are created. The uncoupling of system connections and the introduction of test cabling shall be kept to a minimum. Consideration shall be given to flexible automatic checkout equipment.

3.4.2.10.3 Communication System.- The Apollo Communications System shall provide the following:

Voice Communication
Telemetry
Television
Tracking Transponders
Radio Recovery Aids
Antenna Subsystems
Radar Altimeter (if required by Guidance System)

The following systems descriptions are based upon GOSS utilization employing the currently supported HF, VHF, and C-band frequencies for near-earth communications, and the UHF frequencies for lunar distance communications. However a transition to a single (unified) UHF carrier frequency, using modulation techniques typical of the present DSIF system is contemplated for both near-earth and lunar communications. The system design shall allow provision for this transition.

3.4.2.10.3.1 Voice Communication.- Two-way voice communication capability between the individual crew members, between the spacecraft and earth-based stations, and between each spacecraft in a rendezvous maneuver shall be provided. A personal communication system (NASA-supplied) shall provide two-way voice communication between crew members whether internal or external to the spacecraft. An intercommunication (plug-in) system shall be supplied by the Contractor. Reliable communication in the near-earth phase of flight shall be afforded by a UHF link to that range at which DSIF communications can be acquired and maintained for all potential flight paths. Voice communication using the UHF DSIF transponder shall provide reliable voice transmission and reception to lunar distance.

3.4.2.10.3.2 Telemetry.- A flexible pulse-code-modulation telemetry subsystem compatible with both the VHF and UHF transmission systems shall be provided. Initial telemetry and display system design shall incorporate flexibility to add a ground spacecraft data link.

3.4.2.10.3.3 Television.- A television closed-circuit subsystem for use by the crew in monitoring internal and external scenes in real-time shall be provided. A portable television subsystem capable of real-time and high resolution picture transmission shall be provided. Optimum modulation method shall be employed. Frame rate and resolution trade-offs with transmitter power and antenna size shall be optimized.

3.4.2.10.3.4 Tracking Transponders.- A C-band transponder subsystem compatible with the AN/FPS-16 and equivalent radars shall be provided. This subsystem shall be capable of providing reliable tracking signals in the near-earth-phase of flight to that range at which DSIF tracking can be acquired and maintained for all potential flight paths. A UHF transponder providing reliable velocity and range tracking to lunar distance when used with the DSIF shall be supplied.

3.4.2.10.3.5 Radio Recovery Aids.- The radio recovery aids subsystem shall consist of an HF transceiver system which may be either voice or tone-modulated, and a VHF beacon.

3.4.2.10.3.6 Antennas.- The near-earth antenna system shall consist of multiple flush-mouthed antennas which essentially provide omnidirectional patterns in a plane perpendicular to the booster logitudinal axis. A similar antenna compatible with DSIF shall be used at minor deep-space distances. This antenna shall offer

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sufficient gain for reliable transfer of priority information at a reduced bandwidth in an emergency condition up to lunar distances. The directional antenna system shall be either designed for stresses encountered throughout the mission or be retractable for periods of high stress (such as lunar landing and launch). Both manual and automatic antenna steering shall be provided for the directional antenna.

3.4.2.10.3.7 Radar Altimeter.- A rendezvous and altimeter radar system commensurate with guidance requirements shall be provided by the Contractor if such a system is required.

3.4.2.10.4 Instrumentation.-

3.4.2.10.4.1 General.- The Instrumentation System shall detect, measure and display all parameters required by the crew for monitoring and evaluating the integrity and environment of the spacecraft and performance of the Spacecraft systems. It shall provide data for transmission to earth, to facilitate ground assessment of Spacecraft performance and failure analysis. It shall provide the crew with information as required for abort decision and aid in the selection of lunar landing sites. In addition, the capability shall be provided for documenting the mission through photography and recording.

3.4.2.10.4.2 Measurements.- Prior to procurement of any instrumentation system components, the Contractor shall provide a complete tabulation of all proposed measurements, for review and approval of the NASA (MSC). This tabulation shall include the number and type of all measurements; sensor characteristics; conditions when taken (flight, flight phase, etc.); data disposition, i.e., displayed, real-time telemetry, recorded for telemetry playback, recorded for storage, etc. A block diagram showing the interrelationship of the instrumentation components shall also be provided. Prior to purchase of sensors for flight, type approval will be obtained from MSC.

3.4.2.10.4.3 Sensors.- The sensors selected for each application shall have an inherent reliability at least one order of magnitude greater than the measured and measuring subsystem and shall be compensated such that their capability to perform the intended function is not degraded by the environmental conditions to which they are subjected. Excitation voltage, where required, shall be standardized. Transducer output shall also be standardized, insofar as practicable. Inaccessible measurement areas shall be provided with both primary and spare sensors and associated auxiliary equipment as required. Electrical leads associated with the sensors shall be electrically shielded and mechanically secured so as to minimize the generation or pickup of noise by the leads.

3.4.2.10.4.4 Data Disposition.- The capability shall be provided for data transmission upon crew command or onboard programmed command (e.g., five minutes transmission each hour during the coast phase of the mission). Provisions shall be made for transmitting data in the following critical areas:

a. Measurements relating directly to conditions having an immediate effect on crew safety.

b. Sufficient measurements in each functional area or system to facilitate failure analysis in event of an unsuccessful flight.

c. Navigation and guidance data as required to permit ground station checking of vehicle position and course.

3.4.2.10.4.5 Tape Recorders.- One recorder system shall be provided for the storage of telemetry, voice, and possibly video information for later playback at the discretion of the crew, or for data storage pending spacecraft recovery. A second recorder system shall be provided for multichannel recording of high-frequency parameters such as sound and vibration. This recorder shall also be suitable for use in conjunction with the scientific and biomedical instruments.

3.4.2.10.4.6 Panel Display Indicators.- The panel display indicator shall not be coupled directly to those data channels which are providing similar information to the telemetry or recording system, i.e., there shall be no coupling between the panel display instruments and telemetry/recording channels which could result in cross effects between the circuits, even in event of malfunction.

3.4.2.10.4.7 Calibration.- A calibration feature shall be provided as an integral part of the measurement system and shall be such as to provide a rapid analytic assessment of the measurement system's performance. The method of calibration shall encompass the overall system where practicable, and in addition shall include selectivity of automatic or manual operation at the crew's discretion.

3.4.2.10.4.8 Clock.- Redundant, real-time, binary code generating devices shall be provided to act as the primary time reference; to correlate all data; and to function as an integral part of all time-critical operations. The accuracy and stability of the clock under the environmental conditions expected shall be compatible with the navigation and guidance requirements and future scientific needs.

3.4.2.10.4.9 Telescope.- A gimbal-mounted telescope shall be provided to aid in visual study and photography of the lunar surface and celestial bodies. Dual operating modes shall be possible (high power - narrow angle field of view, or low power - wide angle field of view). Reference axis information shall be provided.

3.4.2.10.4.10 Cameras.- A mounted telescope with a directable field of view shall be provided for lunar surface exploration: one suitable for high-resolution stills; the other suitable for general motion photography of the lunar expedition. Two onboard time correlated cameras shall be used on board; one suitable for monitoring the crew, displays, and spacecraft interior; the other suitable for lunar photography and stellar studies. The latter camera shall be capable of use in conjunction with the telescope or independent use at the crew's discretion.

3.4.3 Lunar Landing Module Systems. - The Spacecraft includes a Lunar Landing Module during lunar-landing missions to provide the gross velocity increment to brake into a lunar-orbit and thence retrograde, descend, hover, and touchdown on the lunar surface; and, to provide a base from which the Command and Service Modules launch themselves from the lunar surface. The Lunar Landing Module configuration is shown on Figure 57. Detail features are not intended to be inferred by the figure.

3.4.3.1 Lunar Touchdown System. - The Lunar Landing Module incorporates a Lunar Landing Touchdown System to arrest impact, support the Spacecraft during its period on the moon, and provide a launching base.

3.4.3.1.1 Structural Load Paths.- The Lunar Touchdown System shall be designed to take advantage of existing structural arrangements provided for tankage and general propulsion systems. Hard points are to be provided on the Lunar Landing Module which will accommodate variations in landing gear geometries and have load distribution capabilities compatible with anticipated landing gear loads.

3.4.3.1.2 Interference with Other Systems.- The Lunar Touch down System shall be designed such that in all positions of stowage and deploymnet it does not interfere with the use of the propulsion systems for midcourse guidance or attitude control purposes.

3.4.3.1.3 Stowability.- The portions of the Lunar Touchdown System which are stowed external to the structural shell are to be faired to minimize aerodynamic affects.

3.4.3.1.4 Deployment.- Deployment may be performed manually by crew in extra-vehicular suit.

3.4.3.1.5 Inspection Maintenance and Servicing. - The design shall be such that advantage can be taken of the crews capabilities in extra vehicular suits both in flight and on the lunar surface, for inspection, maintenance, and servicing. Crew participation is to be considered where significant advantage is attained in reducing complexity and improving reliability without undue demands on the crew's capabilities.

3.4.3.1.6 Post Landing Attitude. - The Spacecraft inherently shall return to or be in a near vertical condition satisfactory for lunar launch or normal egress.

3.4.3.1.7 Surface Condition . - The Lunar Touchdown System shall cater for landings on maria surfaces or crater floors as described in the lunar surface criteria.

3.4.3.2 Guidance and Control System. - The Command Module Guidance and Control System provides all such functions required of the Spacecraft.

3.4.3.3 Reaction Control System. - The Reaction Control System in the Service Module provides all such functions required of the Spacecraft.

3.4.3.4 Propulsion System. - The Lunar Landing Module utilizes a composite propulsion system comprising of multiple lunar retrograde engines for the gross velocity increments required for lunar-orbiting and lunar-landing; a lunar-landing engine for velocity vector control, midcourse velocity control and the lunar hover and touchdown maneuver.

3.4.3.4.1 Requirements. - The system satisfies the following requirements.

3.4.3.4.1.1 Translunar. - The velocity increments required for translunar midcourse control will be provided by the landing engine.

3.4.3.4.1.2 Lunar Orbit Injection. - The velocity increment required for injection into lunar orbit for landing reconnaissance will largely be provided by the retrograde engines. Velocity vector control and vernier velocity control will be provided by the landing engine.

3.4.3.4.1.3 Lunar Orbit Retrograde.- The lunar retrograde engines will largely provide the velocity increment required for retrograde from lunar orbit and subsequent descent to a discreet point above the landing site and with a Spacecraft attitude compatible with the terminal landing maneuver. Velocity vector control and vernier velocity control will be provided by the landing engine.

3.4.3.4.1.4 Hover, Descent, Translation and Touchdown Mode.- The lunar landing engine must provide the capability for hover, descent, and translation involved in touchdown maneuver.

3.4.3.4.1.5 Attitude.- The landing engine must be capable of maintaining the attitude of the Spacecraft compatible with safe abort during all phases of the landing maneuver.

3.4.3.4.2 Description.-

3.4.3.4.2.1 Lunar Retrograde Engines.- The lunar retrograde engines utilize liquid oxygen and liquid hydrogen as propellants. The engines should be pressure fed and sized to provide optimum thrust to weight ratio during the lunar orbit retrograde maneuver giving consideration to the limitations imposed by manual control at the option of the crew.

3.4.3.4.2.1.1 Control.- The lunar retrograde engines shall permit the choice of automatic or manual override of major engine functions.

3.4.3.4.2.2 Lunar Landing Engine.- The single lunar landing engine will utilize the same propellant supply as the lunar retrograde engines and is throttlable over a ratio of $\pm$ 50% about the nominal value. The nominal value shall provide for steady state hovering. The engine shall be capable of multiple starts within the design operating life of the engine.

3.4.3.4.2.2.1 Control.- The Terminal Propulsion System shall provide the choice of automatic or manual override of major engine functions.

3.4.3.4.3 Reliability.- In the event that redundant engine or engines are employed to insure the required reliability, the engines will not be started until required in order to maintain a nearly constant thrust level and landing within or near a preselected area. Velocity vector control will be obtained by gimbaling the landing engine. Each of the main propulsion engines shall be capable of at least three starts.

3.4.3.5 Structural System.- Primary structure shall be designed and evaluated in accordance with standard aircraft practice and by the same criteria as are the Command and Service Modules. The Structural System shall include meteoroid protection, radiation protection inherent in the Structure and Thermal Protection Systems.

3.4.4 Space Laboratory Module Systems.- The Space Laboratory Module is required by the Technical Guidelines to provide its own power supply, environmental control, stabilization, etc., without demand upon the Command and Service Modules' Systems.

3.4.4.1 Reaction Control System.- The Reaction Control System aboard the Service Module will satisfy presently-defined attitude stabilization requirements of the Space Laboratory while they are joined. Until more definitive requirements are specified, the Technical Guideline covering this point will be waived and Space Laboratory Attitude Stabilization will be provied by the Service Module.

3.4.4.2 Structural System.- The Space Laboratory Structural System shall meet the requirements as specified for the Command and Service Modules.

3.4.4.3 Environmental Control System.- The Space Laboratory includes an Environmental Control System to provide a "shirtsleeve" environment in the Space Laboratory and thermal control of Space Laboratory equipment mounted outside of the pressure cabin.

3.4.4.3.1 Requirements. - The system satisfies the following requirements.

3.4.4.3.1.1 Metabolic Requirements.- Metabolic requirements of two of the three man crew, except for their food and water which is supplied by the Command Module.

3.4.4.3.1.2 Equipment Cooling.- Thermal control of equipment.

3.4.4.3.2 Description.- The Environmental Control System is the same as used in the Command Module and as diagrammed in Figure 76.

3.4.4.3.3 Controls.- Control of atmospheric pressures, humidity, and temperature is automatic with provisions for manual adjustment and control.

3.4.4.4 Electrical Power Supply System.- The Space Laboratory includes an Electrical Power Supply System to provide, regulate, and distribute all electrical power required by the Space Laboratory.

3.4.4.4.1 Requirement.- The system provides a continuous power level of 1.5 KW for the nominal sunlit and shadow cycle. For special modes, the system shall supply an average of 1.0 KW for up to three hours continuous shadow operation.

3.4.4.4.2 Description.- The system consists of a solar cell array for primary power and a nickel-cadmium battery for secondary power. Voltage and control is compatible with the Command and Service Modules Systems.

3.4.4.4.3 Control.- System operation is automatic. The crew will monitor and perform switching operations to distribute the power to the proper busses.

3.5 MISSION CONTROL CENTER (MCC) AND GROUND OPERATIONAL SUPPORT SYSTEM (GROSS).- The design configuration of the Mission Control Center and the ultimate design configuration of the Ground Operational Support System have not yet been established. This section describes the operational concept for the Mission Control Center and the computational facilities, gives a general description of the initial configuration of the Ground Operational Support System, and a general outline of the ultimate configuration as currently visualized. The contractor shall ensure that the Spacecraft design and the Ground Operational Support Systems are fully compatible for all phases of manned and unmanned flights. Existing facilities such as the Mercury network and the DSIF, both appropriately modified, will probably be used in the Apollo program.

3.5.1 General Description.- Overall control of all Apollo support elements throughout all phases of a mission will be accomplished from a Mission Control Center (MCC). Mission launch activities up to the time of liftoff will be conducted from a launch control center, at Cape Canaveral. In addition to the launch control center, two types of remote stations will be used. The first type of station will provide support for the following communications: voice, telemetry reception and data processing, data transmission from the ground to the Spacecraft, tracking to determine Spacecraft position and velocity with appropriate data processing and an acquisition system for antenna pointing. Initially the ground support for earth-orbital flights will be supported by modified Mercury-type stations to support the diversified use of frequencies in the VHF and C-Bands, with the contemplated use of the DSIF UHF frequency band for lunar distance communications. It is visualized that eventually some of the Mercury-type sites will be modified or new sites will be implemented to operate using a unified UHF frequency which will support all voice, telemetry, television, and ranging information for near-earth and lunar distances. In the proposed unified concept antenna changes will be required at the existing Mercury sites to enable those sites to have both near-earth and lunar distance capabilities. However, the DSIF would have no role in the near-earth and earth-orbital missions. The second type of remote station will be equipped for use in tracking the Command Module during reentry. Some new stations will probably be required, particularly to provide tracking during reentry. These stations will be located both on land and on ships. The remote stations will be connected to the communications and computation centers located in the Mission Control Center by landlines, submarine cables, and/or by radio, depending on the location of the remote stations. A station will be located at the NASA-Manned Spacecraft Center at Houston. This station may be used for Spacecraft-G.O.SS equipment compatibility checks, simulated missions, astronaut-ground procedure training, development of network operational procedures, as well as for actual missions.

3.5.1.1 Mission Control Center.- The Mission Control Center will have the capability of monitoring the Spacecraft and directing the support elements for all phases of Apollo missions including unmanned and manned earth-orbital and translunar flights. In addition, this control center will include the simulation and training facilities associated with the Spacecraft and the GOSS. These simulation and training facilities include the Apollo crew trainer coupled into the Mission Control Center and simulated remote stations located adjacent to the Mission Control Center. This will combine crew and Mission Control Center personnel training and procedural development.

The Mission Control Center will include a communications center and the appropriate data processing equipment and displays required to allow complete and adequate control of the mission information flow. The Mission Control Center will also include the computing facilities that will be required to handle the data processing required for the determination of vehicle position and velocity and other associated computations throughout the flight phase of the mission.

The information flow associated with the Mission Control Center will include voice communication capability to and from the Spacecraft via relay through any of the remote stations, the transmission of information to the Spacecraft (this may be direct commands for unmanned Spacecraft or for assimilation by the crew of manned Spacecraft), the reception of processed vehicle telemetry data from the GOSS stations, the control and conduct of the GOSS, and all tracking and tracking support data.

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3.5.1.2 Launch Control Center.- Launch control, including space vehicle preparation and checkouts, and the launch countdown will be conducted from Cape Canaveral. Launch control center facilities include the displays and communications necessary for monitoring of the progress of the mission countdown and powered flight phase.

3.5.1.3 Remote Station.- The ground transmitted signal will include voice (relayed from MCC if desired), coded information and beacon or transponder interrogation. The Spacecraft transmission and subsequent ground systems reception will include voice, down telemetry, and beacon or transponder response. It is anticipated that as the program advances the RF link between the Spacecraft and the GOSS may transgress from initial HF, VHF and C-Band links to a single UHF link. The basic site equipment includes an automatic data processor which controls the information flow through the site. Where suitable point-to-point communication circuits exist, the voice information is relayed to the MCC. The telemetry and tracking information is recorded on tape units and simultaneously fed into the data processor where the information is selected by established ground rules and/or MCC directives.

The site data processor, in addition to accepting and reprocessing telemetry and tracking data, also deals with information received from the MCC and from the local inputs and delivers it to acquisition consoles, TTY and higher-speed data transmission systems, and to local visual displays.

Acquisition methods and aids will be provided for acquiring the Spacecraft in angle, frequency, and range. Simulation aids for local training and exercises will be provided for the RF systems and for the communication components. Each of the remote ground stations with the exception of the reentry may eventually be capable of a significant communications and tracking capability at lunar distances. Thus, loss of any station is not catastrophic since alternative lunar capability will exist at other stations.

An interim period will exist before the achievement of the contemplated GOSS for Apollo from the current ground equipment employing Verlort and AN/FPS-16 radars, FM/FM VHF telemetry and UHF voice. The transition from the Mercury-type of equipment to the unified frequency system-equipment should be made consistent with the Apollo

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systems development and the Apollo program schedules. The current system will remain substantially unchanged if project schedules for the early Apollo vehicles must carry extensions of the Mercury system.

3.5.1.4 Remote Station Equipment.- Current equipment plans for each ground station includes the use of UHF and VHF communications systems, the use of C-band equipment for near-earth tracking, and the use of UHF transponders for tracking at lunar distances. The unified RF systems concept utilizes a single UHF frequency for both near-earth and lunar communications and tracking.

The unified systems concept requires that the RF modulation techniques used in a single RF UHF carrier bandwidth will be the same for both earth-orbital and lunar missions. The remote station configuration provides an antenna installation supporting two RF information channels (not widely separated) in the same frequency band. These information channels would be used for position and velocity determination, voice, and telemetry. The functional layout of the remote site is shown in figure 76, and figure 77 indicates the type and kinds of data flow that occur in the site.

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3.6 Engineering and Development Test Plan.-

3.6.1 Introduction and Development Philosophy.- The development program required to accomplish the Apollo objective comprises five major phases. These are:

3.6.1.1 Design information and development tests.-

3.6.1.2 Qualification, reliability and integration tests.-

3.6.1.3 Major ground tests.-

3.6.1.4 Major development flight tests.-

3.6.1.5 Flight missions.-

This program presents a normal progression in the develop ment of the components, subsystems, and systems of the Apollo Spacecraft from contractual go-ahead to lunar mission accomplishment.

a. Reliability.- Reliability is of primary importance and is to be emphasized throughout the Apollo Program.

b. Preparatory Testing.- The stringent standards for the manned Apollo missions and the high cost and limited availability of Saturn and Nova Launch Vehicles make it particularly imperative that extensive preparatory ground and development flight testing be conducted.

c. Early Man-Rating of Launch Vehicle and Spacecraft.- The combination of Launch Vehicle expense and schedule, specified Apollo milestones, and "man-in-the-loop" design demands early man-rating of the Space Vehicle. Program emphasis is placed on the early development and qualification of the Launch Escape System.

d. Backup Missions.- Backup missions may be required in order to accomplish objectives not completed on previous missions, and allow investigation of unexpected problem areas in order to prevent or alleviate costly program delays.

e. Buildup Procedure.- Each major advance in the development program should be preceded by a step-by-step buildup in severity of test conditions.

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f. Ground Test and Checkout.- Detailed ground test and checkout of the Apollo Spacecraft are conducted prior to launch. This testing consists of separate module checkout, integrated systems tests, and complete simulated mission testing.

g. Schedule Performance.- Strong emphasis is placed on schedule performance with continuous NASA monitoring and control through the use of NASA PERT.

3.6.1.1 Design Information and Development Tests.- Component and system development tests shall be conducted to provide design and fabrication data to back up engineering specifications. Testing in this category is to demonstrate that each system satisfies design requirements and is a continuation of design requirements and is a continuation of design information testing.

Design information and development tests shall be performed by the Contractor or equipment Sub-Contractor. The most significant design and development tests areas are described in the following paragraphs.

3.6.1.1.1 Wind Tunnel Tests.- Tests to obtain aerodynamic forces, stability data, pressure distributions, and heat transfer data shall be undertaken immediately to acquire essential design information. Subsequent proof tests of final designs and special tests such as aerodynamic noise and buffet determination shall also be conducted. Full utilization of government and private facilities shall be utilized and coordinated with existing and planned NASA research programs. The major tests during the initial stages of design shall determine:

a. Static and dynamic stability of reentry and exit configurations.

b. Stability of the reentry configuration with drogue chute.

c. Reaction jet effects on stability of the reentry configuration.

d. Dynamic stability of the escape configuration.

e. Pressure distributions on the escape, reentry, and exit configuration.

f. Escape rocket jet effects on the escape configuration.

g. Static stability and tower drag of the escape tower alone and in proximity to the Command Module.

h. Self-excited oscillations potentially present on the Command Module in the transonic speed range.

i. Static stability of the Space Vehicle configuration.

j. Pressure distributions on the various launch configurations.

k. Aerodynamic noise and buffet of the launch and escape configuration with tower.

l. Heat transfer on exit and reentry configurations. Extensive testing at ground facilities will be conducted to aid in formulating and verifying calculations of heat transfer in all regimes of flight through parabolic velocity. Consideration of real gas effects, radiative heat transfer phenomena shall be included. These data shall be substantiated further by the reentry flight test program.

m. Landing System forces and stability characteristics.

3.6.1.1.2 Heat Protection.- Tests shall be conducted to develop design and fabrication criteria for ablative and insulative materials. Extensive screening and analysis of candidate heat shield materials shall be conducted to provide early materials selections. Intensive testing and analysis of the selected materials shall be conducted to provide design data and to assure satisfactory thermodynamic performance characteristics during all phases of flight and to verify fabrication methods and structural adequacy. Testing shall utilize the most suitable private and government facilities available to evaluate all significant environmental effects including elevated temperature and heating, vibration, chemical reactions, surface forces, noise, space environment, etc.

3.6.1.1.3 Structure, Materials, and Mechanisms.- Structure and structural elements including panels, joints, attachments, and radiator details shall be developed and tested to qualify designs capable of withstanding loads, temperatures, vacuum, and other environmental conditions. Energy absorption materials and crew support attenuation devices shall be tested to obtain design characteristics. Airlock, interface disconnect, and docking mechanisms developments involve lubricant, sealing, and actuation tests.

3.6.1.1.4 Human Factors Testing.- Tests shall be conducted throughout design and development to demonstrate that the crew can perform with a high level of proficiency, under the anticipated conditions of the mission. Particular attention is directed to man's influence on reliability. Research and testing shall be carried out in the areas of man-machine, man-man, and man-environment interactions.

The major areas to be verified by testing are:

a. Operating procedures for individual Spacecraft systems and for the use of ground support equipment and the Ground Operational Support System during Spacecraft operations.

b. Integrated crew operating procedures for the full mission.

c. Procedures and equipment for in-flight maintenance.

d. Displays and controls, work station arrangement, support and restraint equipment, extra-vehicular viewing equipment and illumination.

e. Environmental sensing and control systems.

To assure both good design inputs and verification of designs, testing shall be continued as equipment and crew requirements are changed. Emphasis during testing shall be placed on performance validation under partial and full mission simulations.

3.6.1.1.5 Cryogenic Storage.- Development tests for cryogenic tankage, heat exchangers and control valving shall be conducted to optimize storage tank design and parameters affecting heat transfer, flow rates, pressure drops and insulating effectiveness.

3.6.1.1.6 Electrical Power System.- Development and qualification tests shall be conducted on the Bacon-type fuel cells, voltage regulator, and potable water separation system. Alternate or backup systems should be investigated.

3.6.1.1.7 Liquid Propulsion Systems.- Development tests shall be conducted on the Liquid Propulsion System including tests of the combustion chambers, throttling, nozzles, fuel distribution systems including valves, filters, and pressurization system. Propellant slosh, vortexing and tank burst-pressure tests shall be conducted. Engine interaction and ignition control system response and effects of rocket engine blast on the lunar surface shall be conducted in hard vacuum tests.

3.6.1.1.8 Solid Propulsion.- Solid rocket igniters and rocket motors shall be tested under environmental extremes to provide confirmation of ignition and exhaust jet wake characteristics, thrust variations, and temperature sensitivities.

3.6.1.1.9 Stabilization and Control.- Tests shall be made of manual control input arrangements along with development of acceptable feel characteristics. Components, circuit designs, and displays shall be tested to develop optimum system design.

3.6.1.1.10 Environmental Control System.- Tests shall be conducted to develop exterior radiator coatings, cryogenic quantity indication, cold plates, catalytic burners.

3.6.1.1.11 Communication and Instrumentation.- Development tests shall be conducted towards improving the reliability and reducing the power requirements of existing communication equipment. Tests shall be conducted to improve the reliability and accuracy of sensing equipment, particularly radiation dosimeters and gas analyzers.

3.6.1.1.12 Parachute Development.- Development tests shall be conducted in the main and drogue parachute system. Parachute deployment techniques shall be investigated and comparative tests conducted.

3.6.1.1.13 Radiation Shielding Development.- Shielding calculations shall be made and continuously revised to take into account new radiation data. Where data is incomplete, NASA will carry out experiments to fill the data gaps. Shield proofing tests shall be carried out on final vehicle assemblies to verify shielding calculations.

3.6.1.2 Qualification, Reliability, and Systems Integration Test.-

3.6.1.2.1 Qualification Tests.- A qualification test program shall be conducted on representative components, subsystems and systems in order to establish the flight worthiness of these components. These tests will be performed at the appropriate facility of the Contractor or Sub-Contractor.

3.6.1.2.2 Reliability Tests.- Tests intended primarily to provide information to corroborate and update reliability estimates shall be performed on prototype and production units. Subsystems and system shall be subjected to accelerated environments and simulated mission duty cycles. Testing priority shall be given to equipment items suspected of being critical from a reliability standpoint. Flight systems and GSE shall be tested at all levels from components through sub-assemblies and complete systems. The number of specimens to be tested will vary with indicated reliability and the results of previous tests.

3.6.1.2.3 Systems Integration and Compatibility Tests.- One complete set of prototype Spacecraft systems shall be programmed early in the development cycle for conducting systems integration and compatibility tests. The GSE made available for this program shall be tested concurrently with the Spacecraft systems. Verification of proper performance shall be made:

a. At the component and subsystem level.

b. At the major system level.

c. Between systems of the same modules; and

d. Between systems of different modules.

Initial determination of systems operational performance and interactions shall be conducted to provide early resolution of compatibility problems and permit prompt design modification.

Systems integration and compatibility testing shall continue throughout the program as changes in system design or components are made.

3.6.1.3 Major Ground Tests.-

3.6.1.3.1 Earth Impact Attenuation Evaluation.- Early confirmation of the earth impact attenuation system and crew shock absorption system shall be conducted on developmental Command Modules. The complete range of land and water impact conditions shall be investigated. Final qualification shall be accomplished on flight hardware.

3.6.1.3.2 Flotation and Egress Tests.- A Command Module structure which reproduces the module geometry, crew arrangement, egress provisions, and mass properties shall be tested to verify module flotation stability and to establish and evaluate crew egress provisions. The requirement for the development of auxiliary module flotation system shall be investigated with this test unit. Crew survival equipment and special procedures for crew safety following water landings shall also be evaluated.

3.6.1.3.3 Human Factors and Bio-Medical Tests.- An early evaluation of the Environmental Control System shall be made. A developmental Command Module will be tested in a space environment simulation chamber. The thermal balance and atmospheric environment shall be evaluated under simulated flight conditions. Additional tests shall evaluate any impairment of crew function resulting from the use of pressure suits. These tests shall also evaluate the mission acceptability, bio-medical instrumentation, sanitation system, duty and rest stations, food and water provisions under simulated space conditions. Crew tasks and displays shall also be appraised in this phase of the program.

3.6.1.3.4 Structural Evaluation.- Structurally complete Command Modules, Service Modules, and Adapters shall be used in the structural test program for testing locally critical conditions, for drop tests, thermal testing and major over-all design conditions. Major design conditions are to be evaluated on an assembled Spacecraft using radiant lamps to simulate aerodynamic heating. Temperature gradients, stresses, and deflections are to be measured and recorded at selected critical locations. Tests of local structure shall be conducted for the most critical load and environment.

Vertical and horizontal impact are to be simulated by drop testing to prove out the impact system and the basic module structure. Appropriate linear and angular accelerations shall be recorded together with loads imposed on the crew restraint system and other critical structure. Anthropomorphic dummies shall be installed to simulate crew restraint loadings.

3.6.1.3.5 Propulsion Systems Static Firing Tests.- Full-scale evaluation of the Service Module Propulsion Systems shall be conducted on a propulsion test article which includes the Propulsion Systems, their associated instrumentation, and Service Module structure.

Rigid mounting fixtures are to be used in performing the Propulsion System tests. Performance tests of the systems shall be conducted.

Test conditions shall include full duration firings, propellant utilization, thrust vector control tests, and evaluation of imposed system malfunction.

System firing tests shall be performed in a low pressure environment to evaluate jet wake interaction and impingement, altitude starting, and for obtaining closer simulation of flight conditions. The starting, normal operation, shutdown transients, and operations under imposed engine and control system malfunctions are demonstrated.

Test firings of the Reaction Control System will be conducted with the Propulsion Systems static firing tests.

3.6.1.3.6 Structural Dynamics.- Vibration and noise tests shall be conducted on a complete spacecraft and on various launch and flight configurations.

3.6.1.3.6.1 Structural Dynamics Tests.- Structural dynamics tests consisting of vibration testing on individual Command and Service Modules to obtain modes of response and amplification factors.

3.6.1.3.6.2 Vibration Tests.- Vibration tests of the combined Spacecraft (Modules, Adapter and Escape Tower) shall be conducted to define the modes and structural amplification factors of internally mounted items. Vibration proof tests with systems operating and monitored will be conducted.

3.6.1.3.6.3 Acoustic Noise Tests.- Acoustic noise tests will be conducted to evaluate the operational and structural integrity of the combined Spacecraft in a high-level noise environment.

3.6.1.3.7 Reliability and Simulated Mission in Space Environment.- Comprehensive ground evaluation of Spacecraft Systems performance shall be performed in a space environment simulator. The space simulator will provide all practicable space environmental conditions including solar radiation, temperatures, heat absorptivity and pressure (hard vacuum). The test program shall cover the Command Module, the Service Module and the complete Spacecraft. Test articles shall have complete prototype systems installed in production structures. Crewman simulators are to be included for test missions requiring metabolic loading. Testing shall progress from short duration periods through complete 14 day tests. Repeated tests are to be performed in order to provide additional component and system reliability.

3.6.1.3.8 Lunar Landing System Simulation.- The criteria for the lunar landing maneuver shall be reviewed and revalued periodically as more data accrue. A comprehensive evaluation of the control and piloting problems involved in a manned lunar landing shall be conducted utilizing analogue computer techniques, dynamically scaled modes earth-based lunar landing simulators and/or simulated lunar landings on earth utilizing tests with mass and performance characteristics similar to the Lunar Landing Module. Dynamic characteristics during lunar impact shall be investigated utilizing dynamically scaled and full-scale models.

3.6.1.3.9 Support Equipment and Launch Complex Compatibility.- A support equipment and launch complex compatibility test Spacecraft shall be provided to develop compatibility of the Spacecraft and its support equipment. The test program includes development of ground support equipment, Spacecraft handling procedures, and operations training.

The test article shall include Command and Service Modules and adapter and is a production configuration with respect to vehicle handling points.

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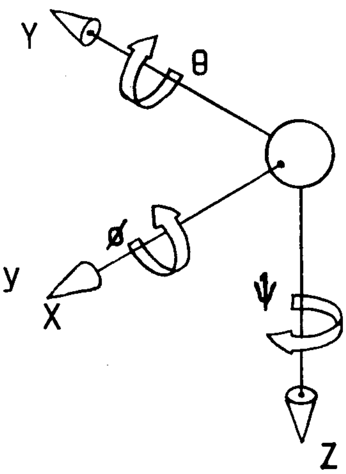
Positive directions of axes and angles (forces and moments) are shown by arrows. (when launch vehicle is at a launch angle of 90°, the positive "X" direction in vertically upwards.)



Axis		Moment about axis		
Designation	Symbol	Designation	Symbol	Positive direction
Longitudinal	X	Rolling	L	Y ⟶ Z
Lateral	Y	Pitching	M	Z ⟶ X
Normal	Z	Yawing	N	X ⟶ Y

Force (parallel to axis) symbol	Angle		Velocities	
	Designation	Symbol	Linear (component along axis)	Angular
X	Roll	ø	u	p
Y	Pitch	θ	v	q
Z	Yaw	ψ	w	r

Figure 1.- Reference axis system.

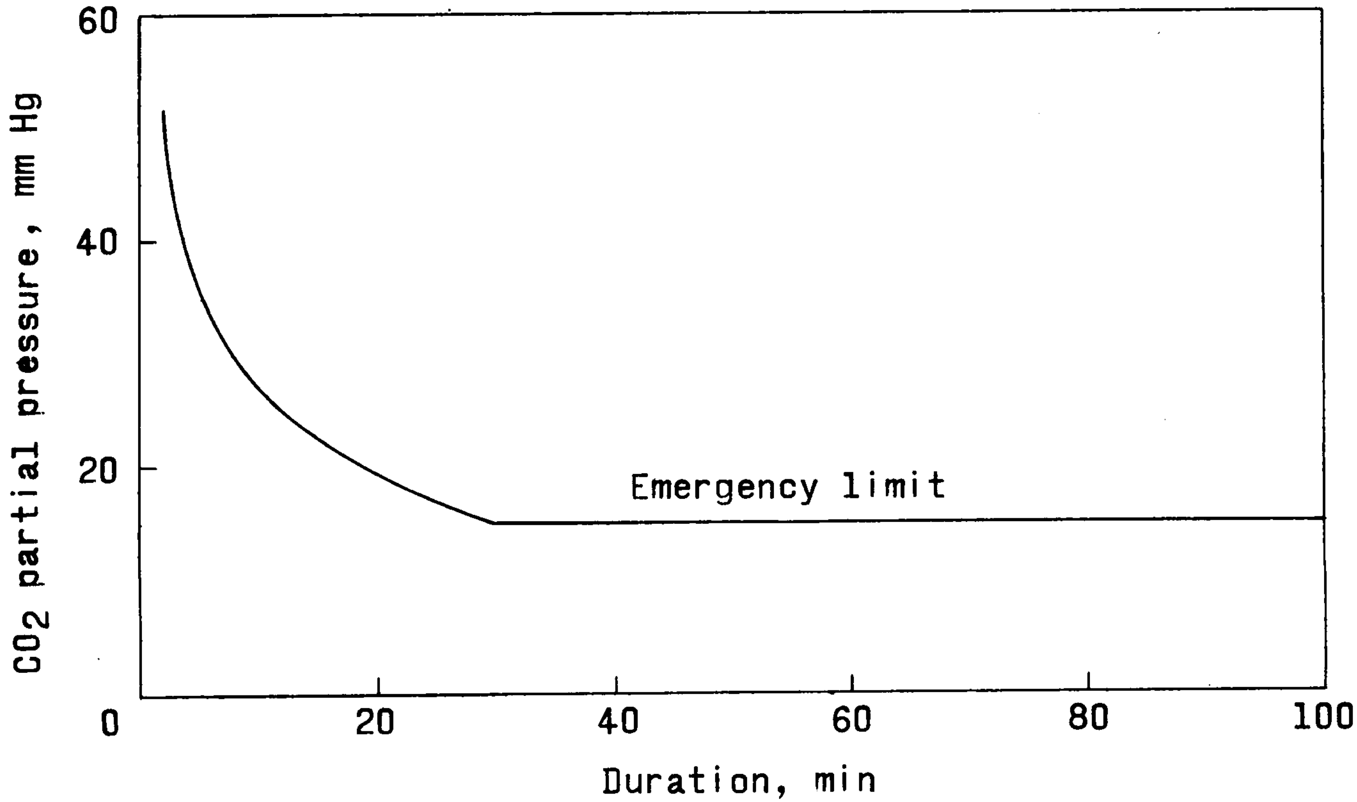


Figure 2.- Emergency carbon dioxide limit.

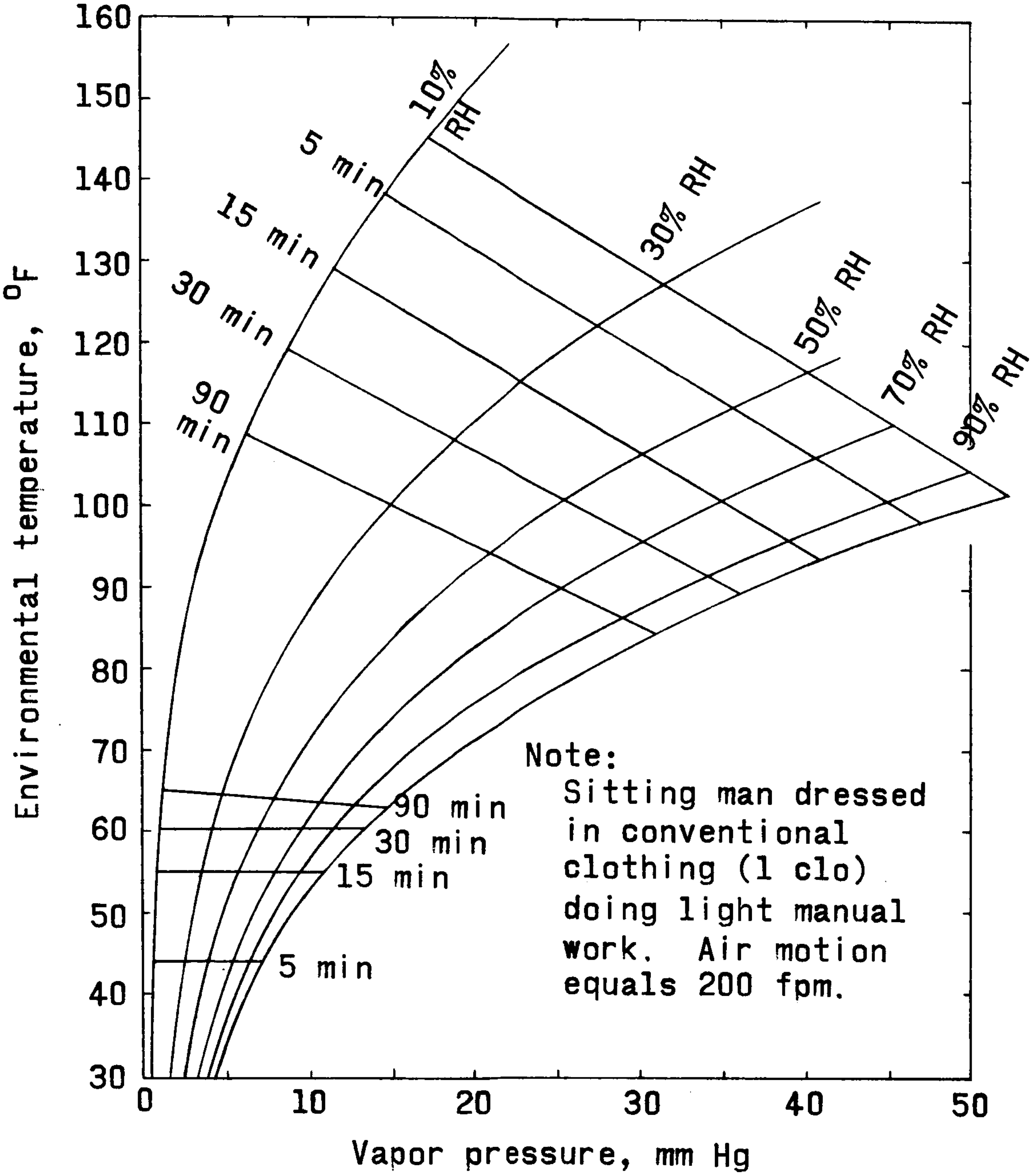


Figure 3.- Temperature and humidity nominal limit.

[redacted]

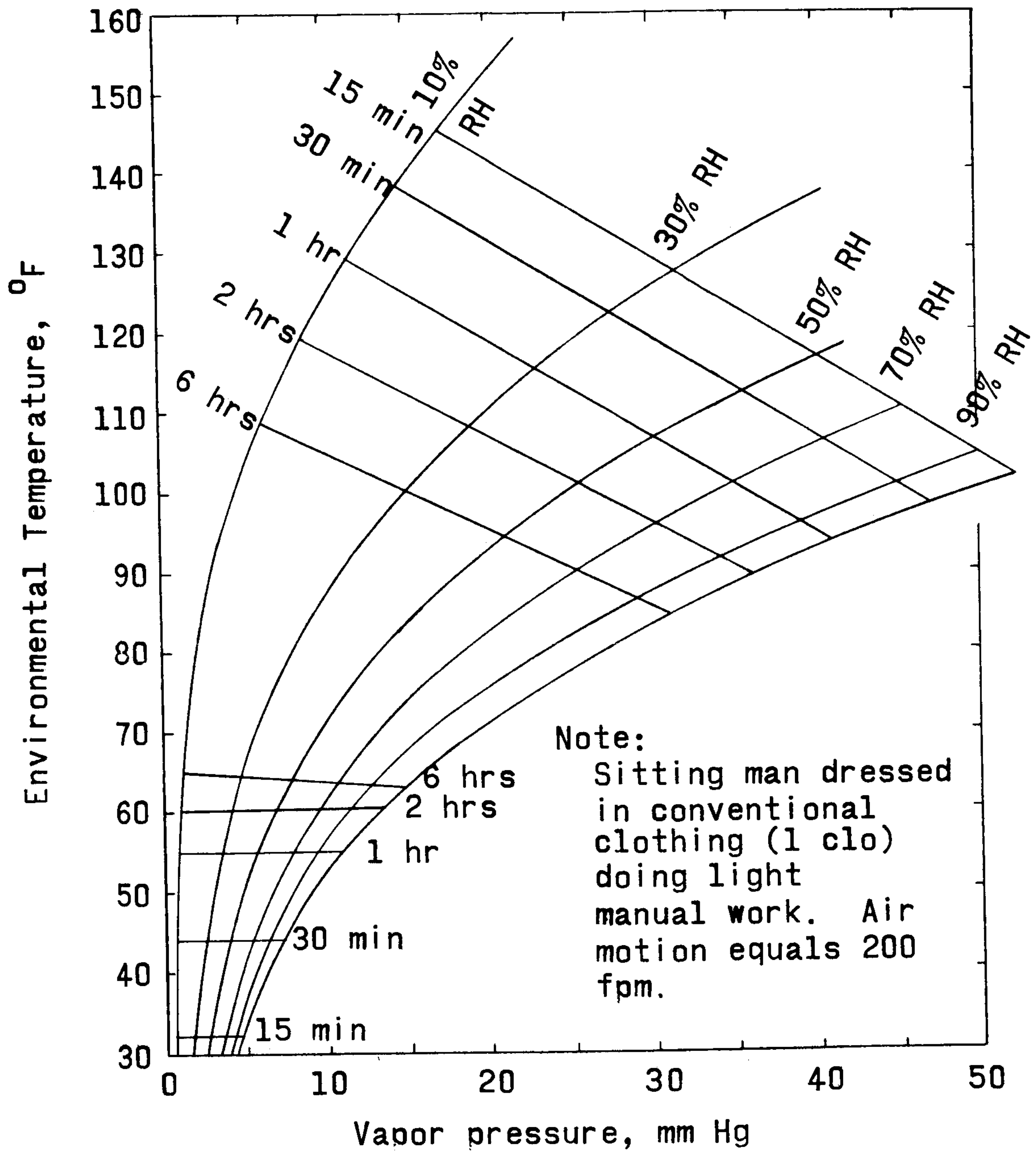


Figure 4.- Temperature and humidity emergency limit.

[redacted]

Critical organ	Maximum permissible integrated dose (rem)	RBE (rem/rad)	Average yearly dose (rad)	Maximum permissible single acute emergency exposure (rad)	Location of dose point*
Skin of whole body	1,630	1.4	233	500[1]	0.07-mm depth from surface of cylinder 2 at highest dose rate point along eyeline
Blood-forming organs	271	1.0	54	200	5-cm depth from surface of cylinder 2
Feet, angles, and hands	3,910	1.4	559	700[2]	0.07-mm depth from surface of cylinder 8 at highest dose point
Eyes	271	2[3]	27	100	3-mm depth from surface on cylinder 1 along eyeline

*See figure 6.

[1]Based on skin erythema level

[2]Based on skin erythema level but these appendages believed to be less radiosensitive

[3]Slightly higher RBE assumed since eyes are believed more radiosensitive

Figure 5.- Radiation exposure dose limits.

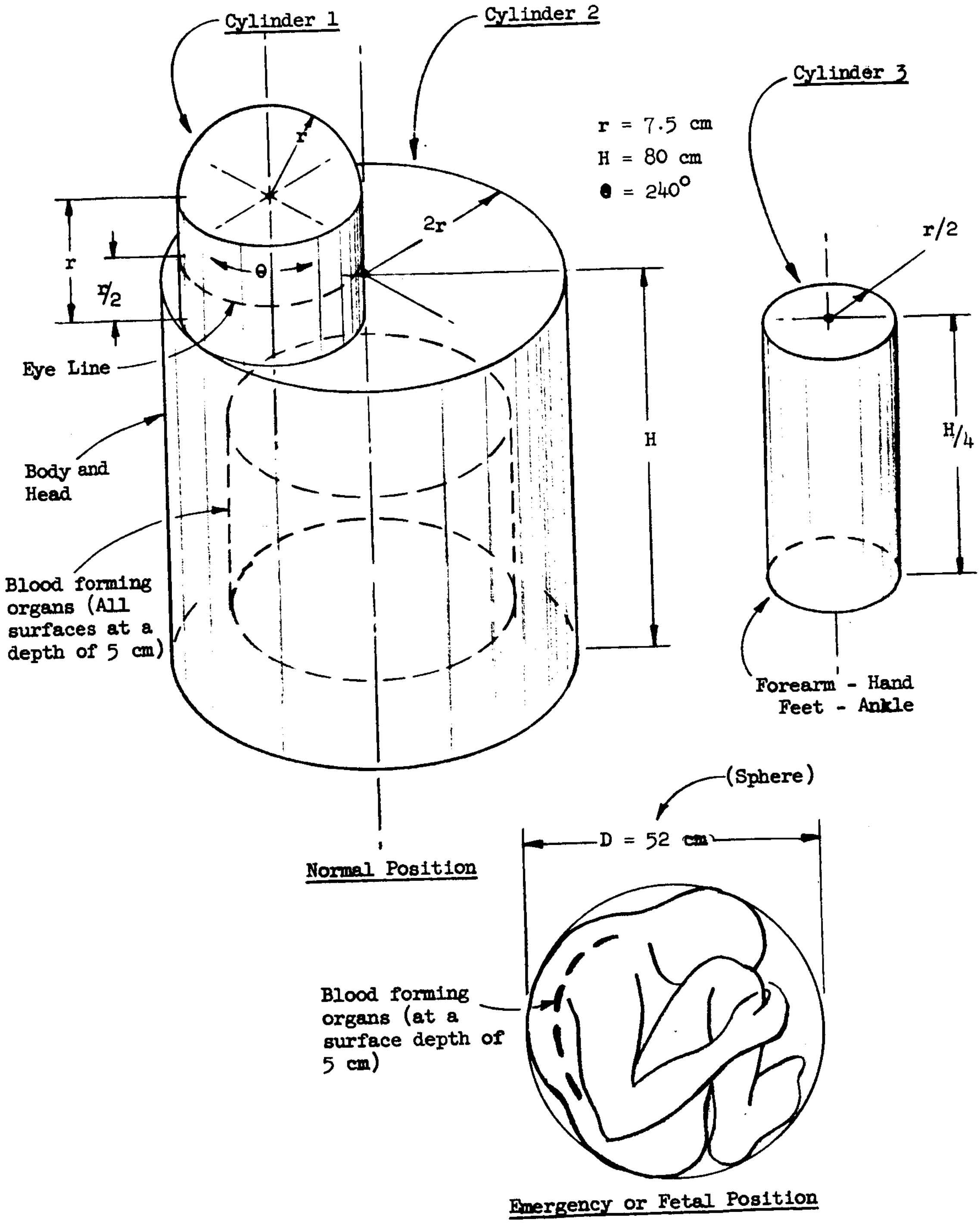


Figure 6.- Models of the radiation standard man.

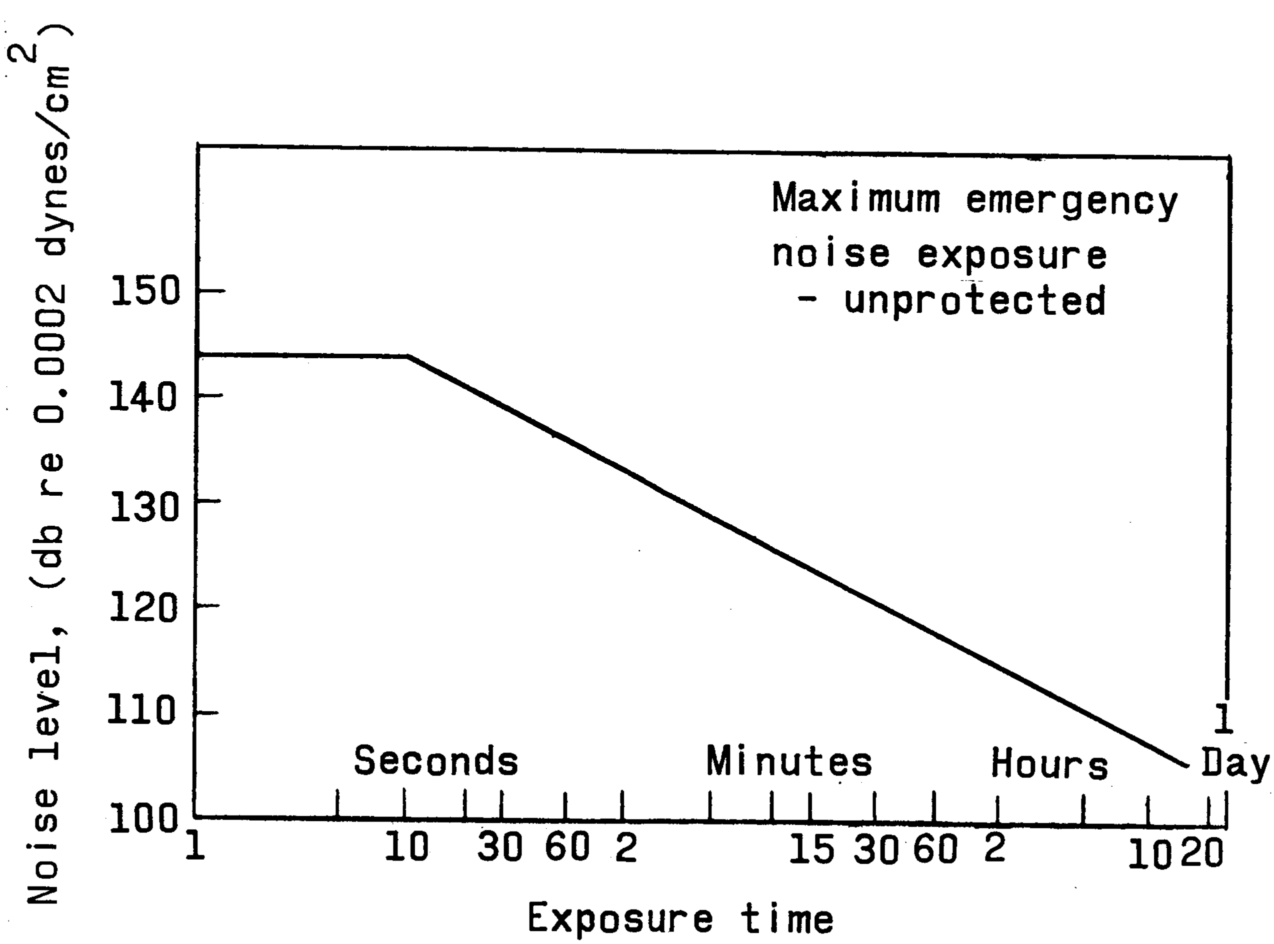


Figure 7.- Noise tolerance, emergency limit.

[redacted]

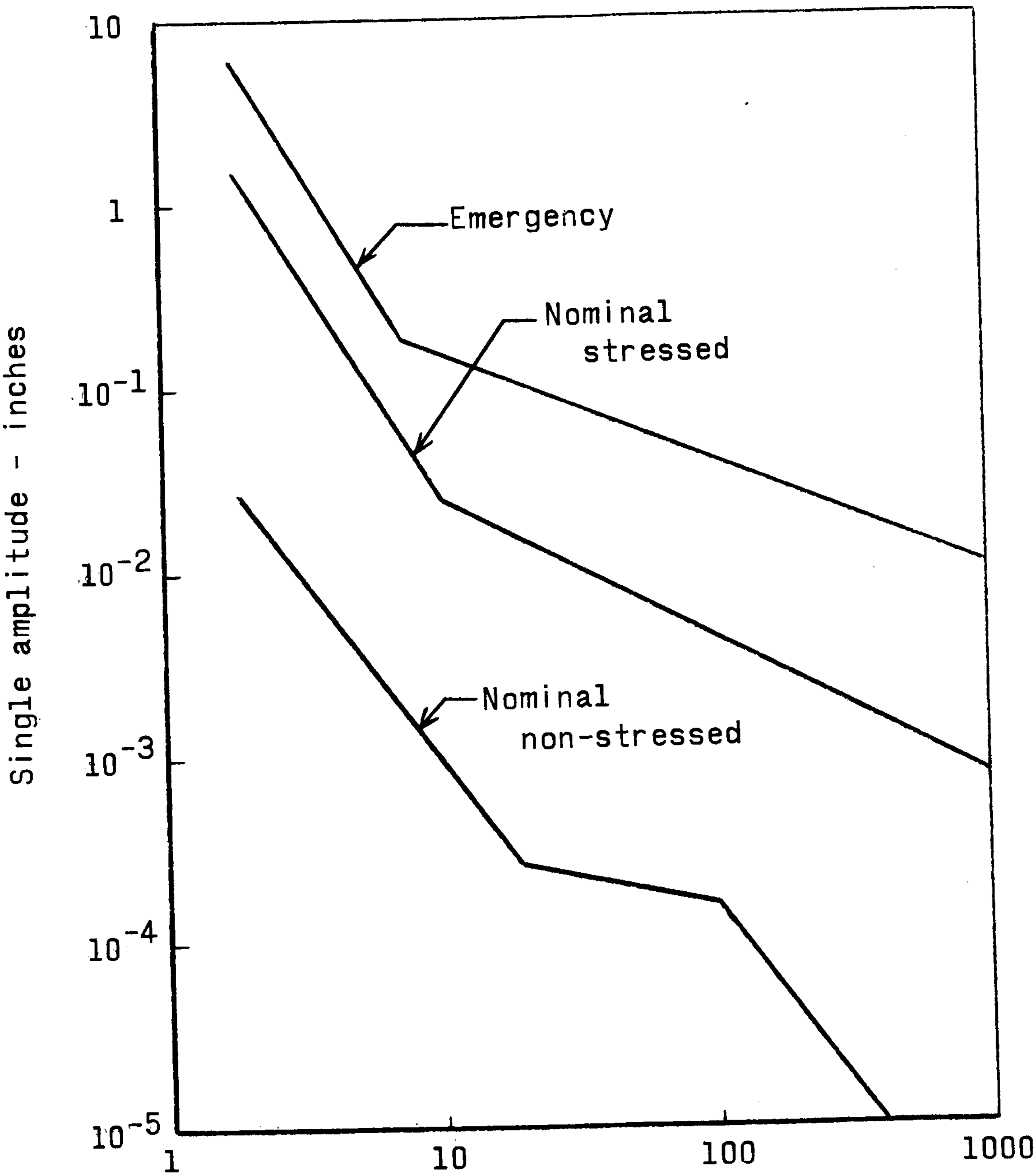


Figure 8.- Vibration limits.

[redacted]

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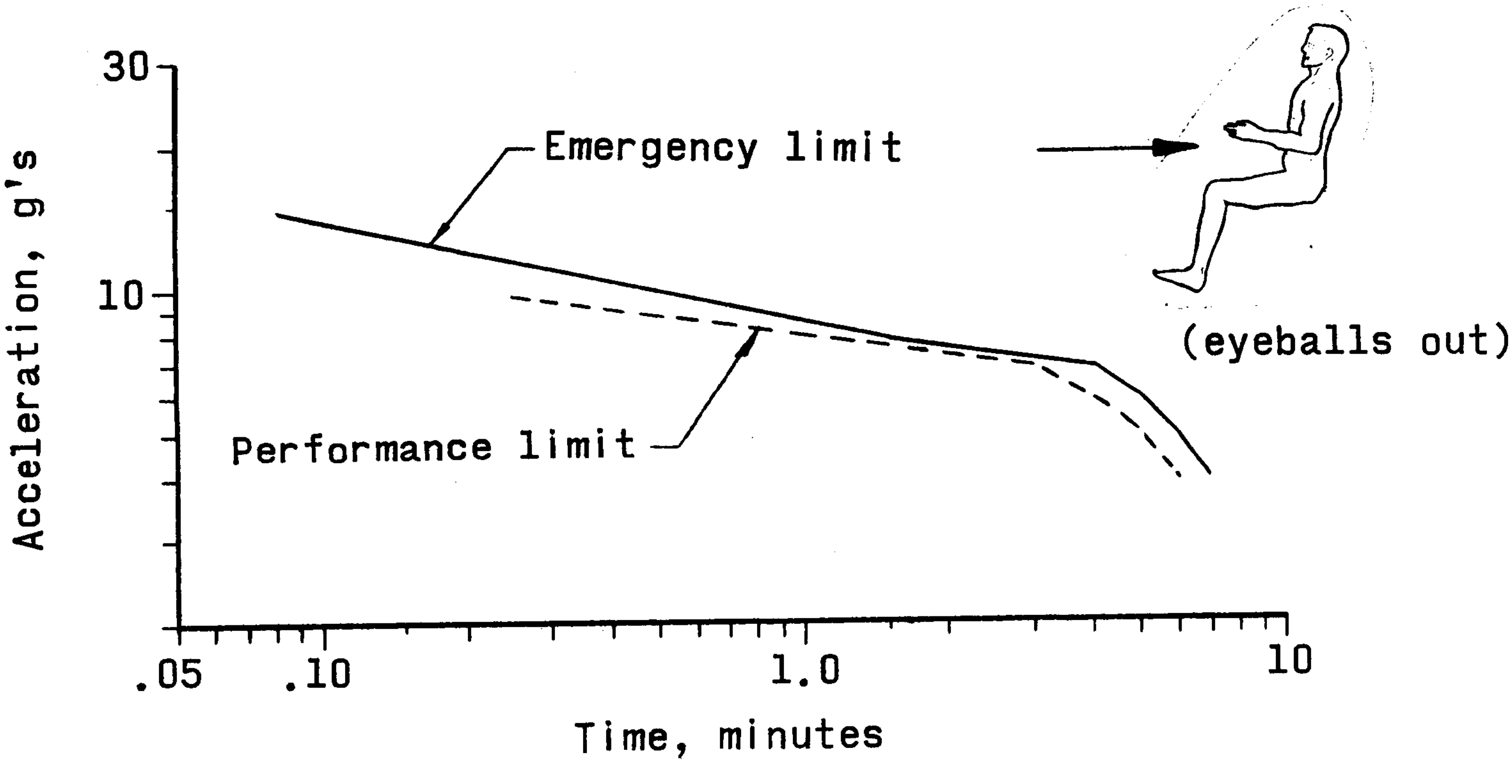


Figure 9.- Sustained acceleration. (References 1 thru 4 and 7).

[redacted]

[redacted]

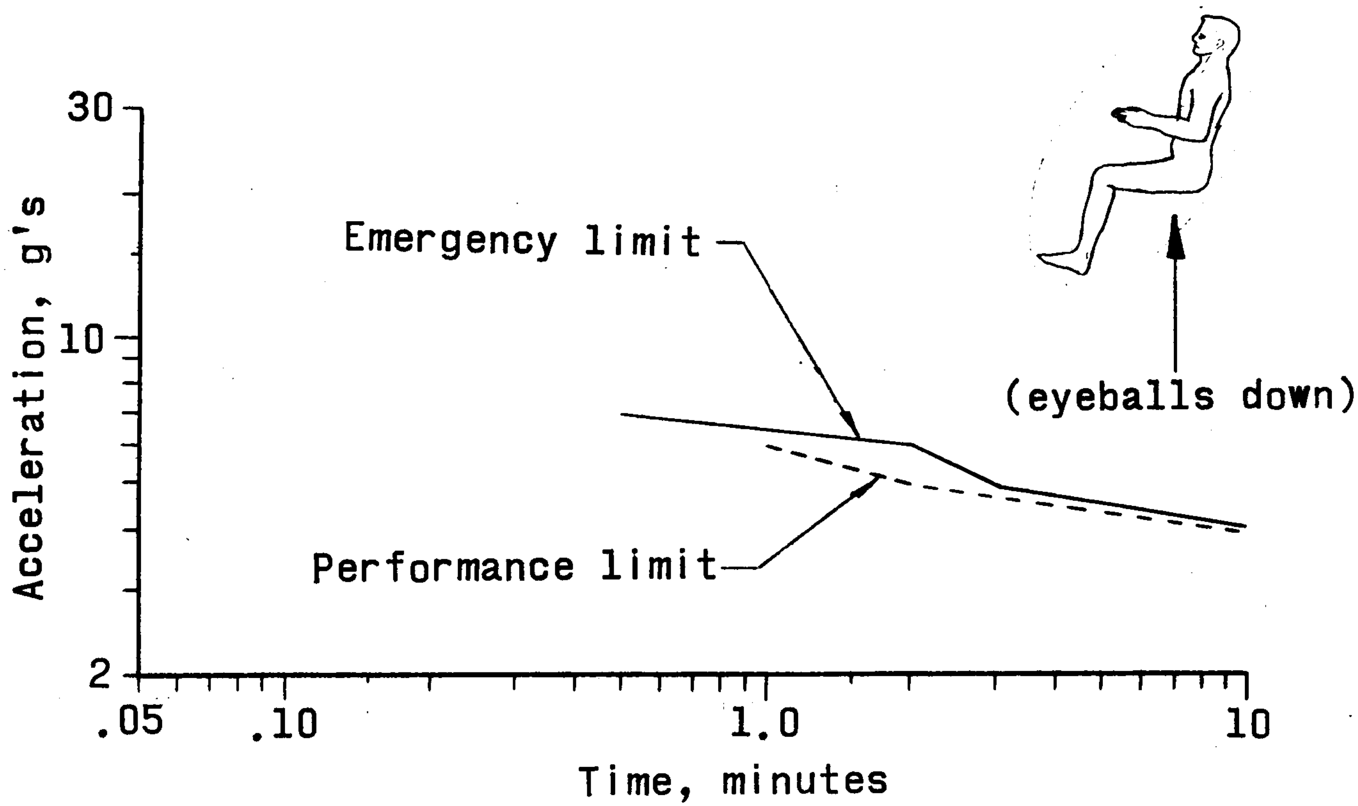


Figure 10.- Sustained acceleration. (References 1, 5 and 6).

[redacted]

[redacted]

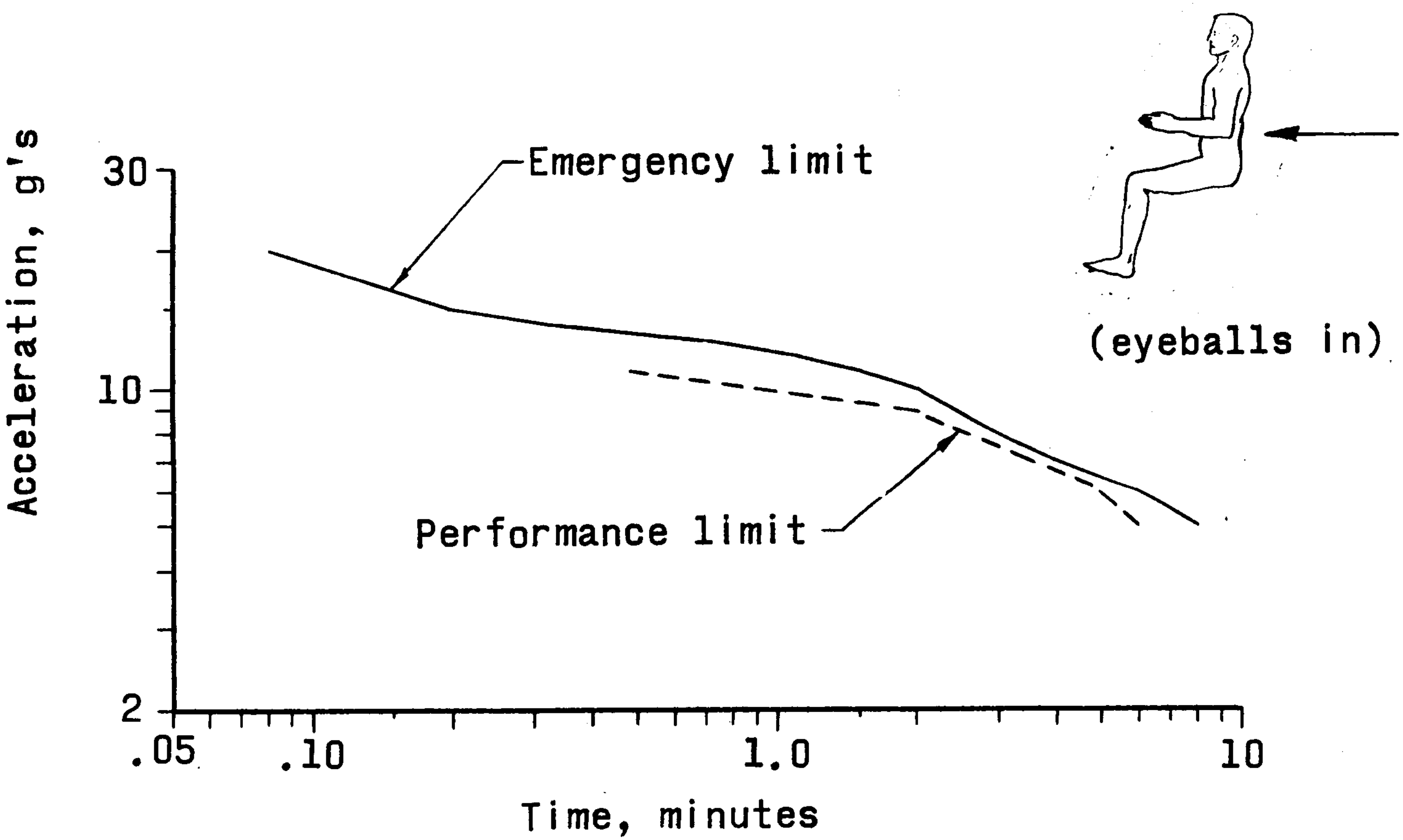


Figure 11.- Sustained acceleration. (References 1 thru 4 and 8).

[redacted]

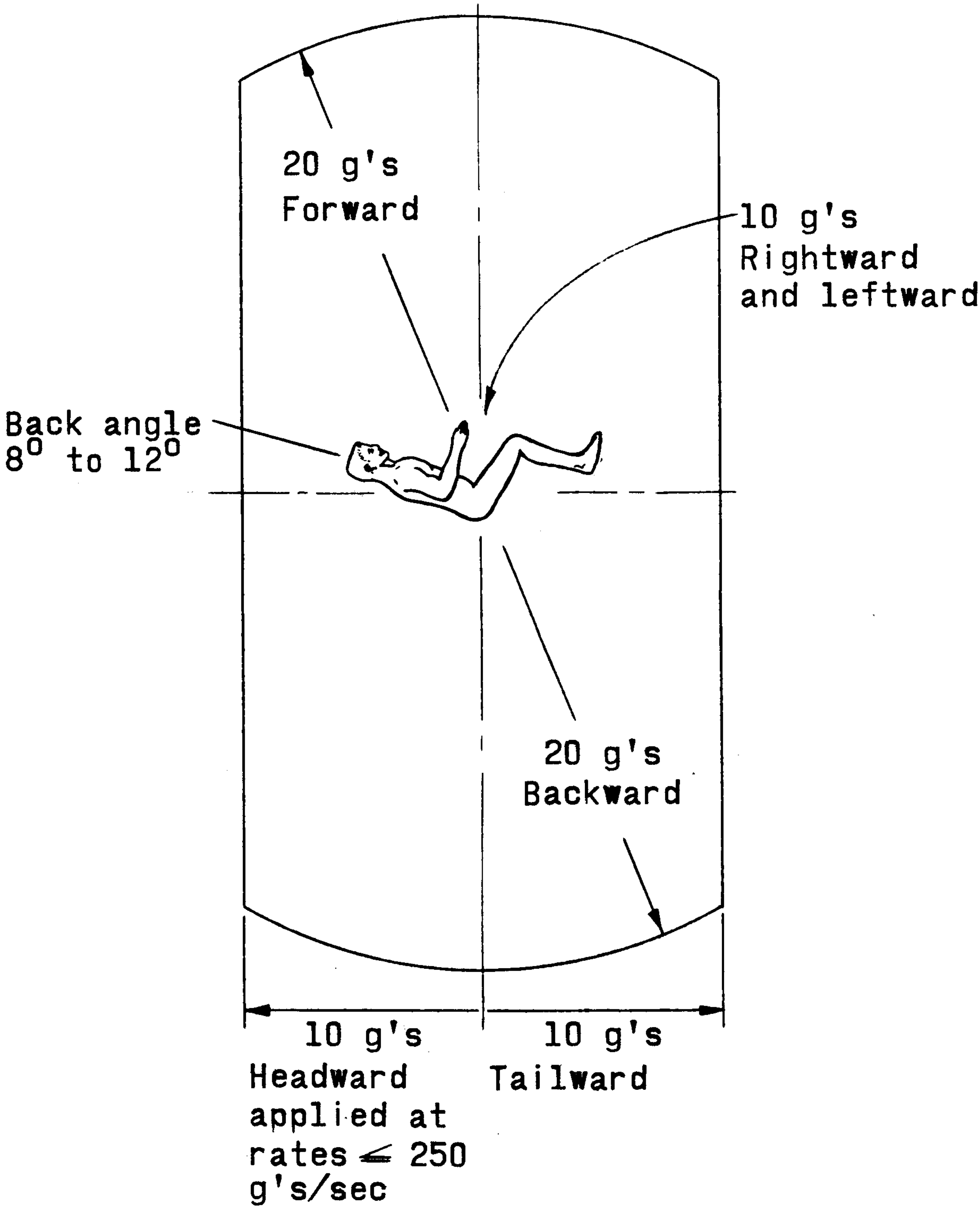


Figure 12.- Impact accelerations - nominal limits.

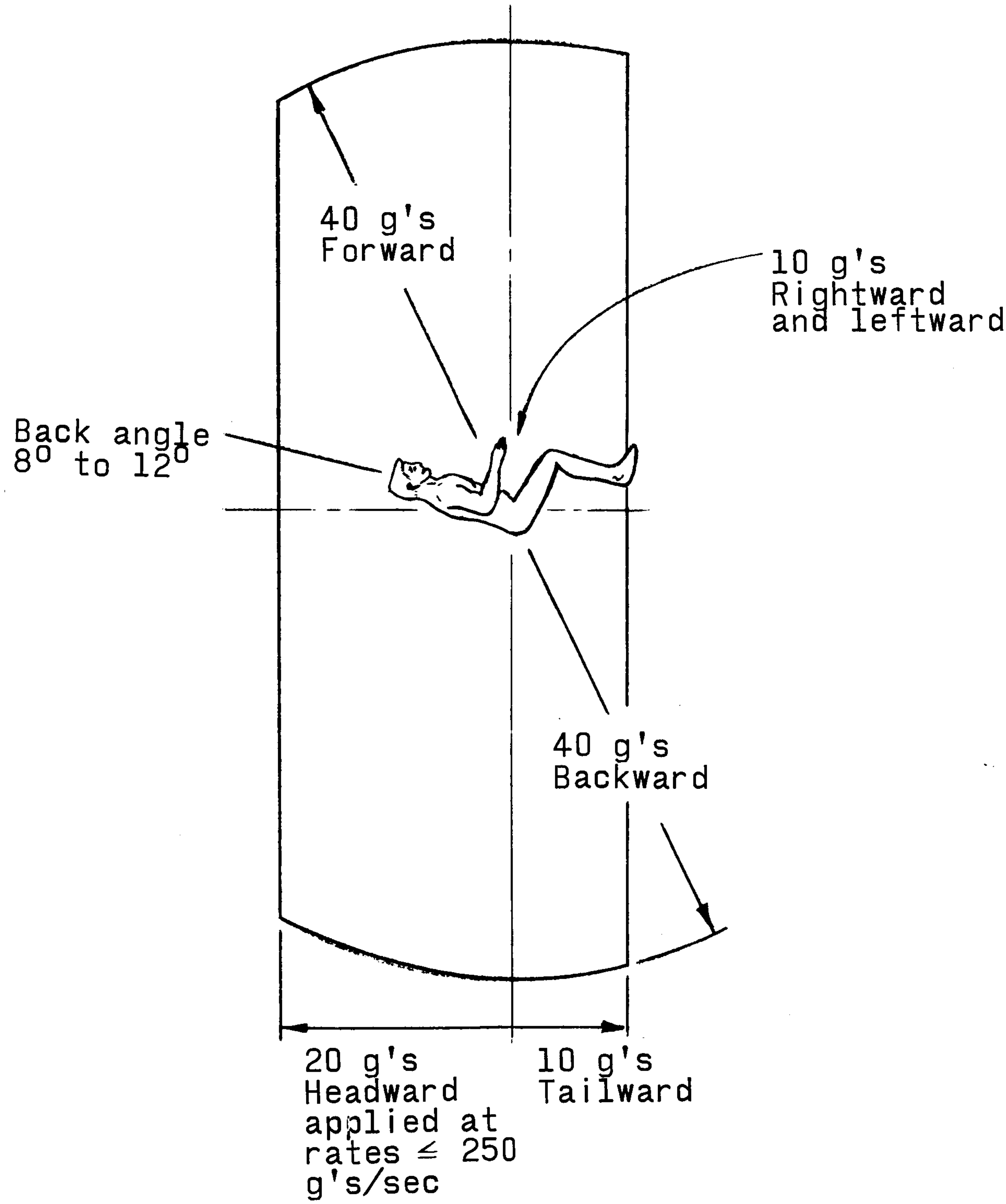


Figure 13.- Impact accelerations - emergency limits.

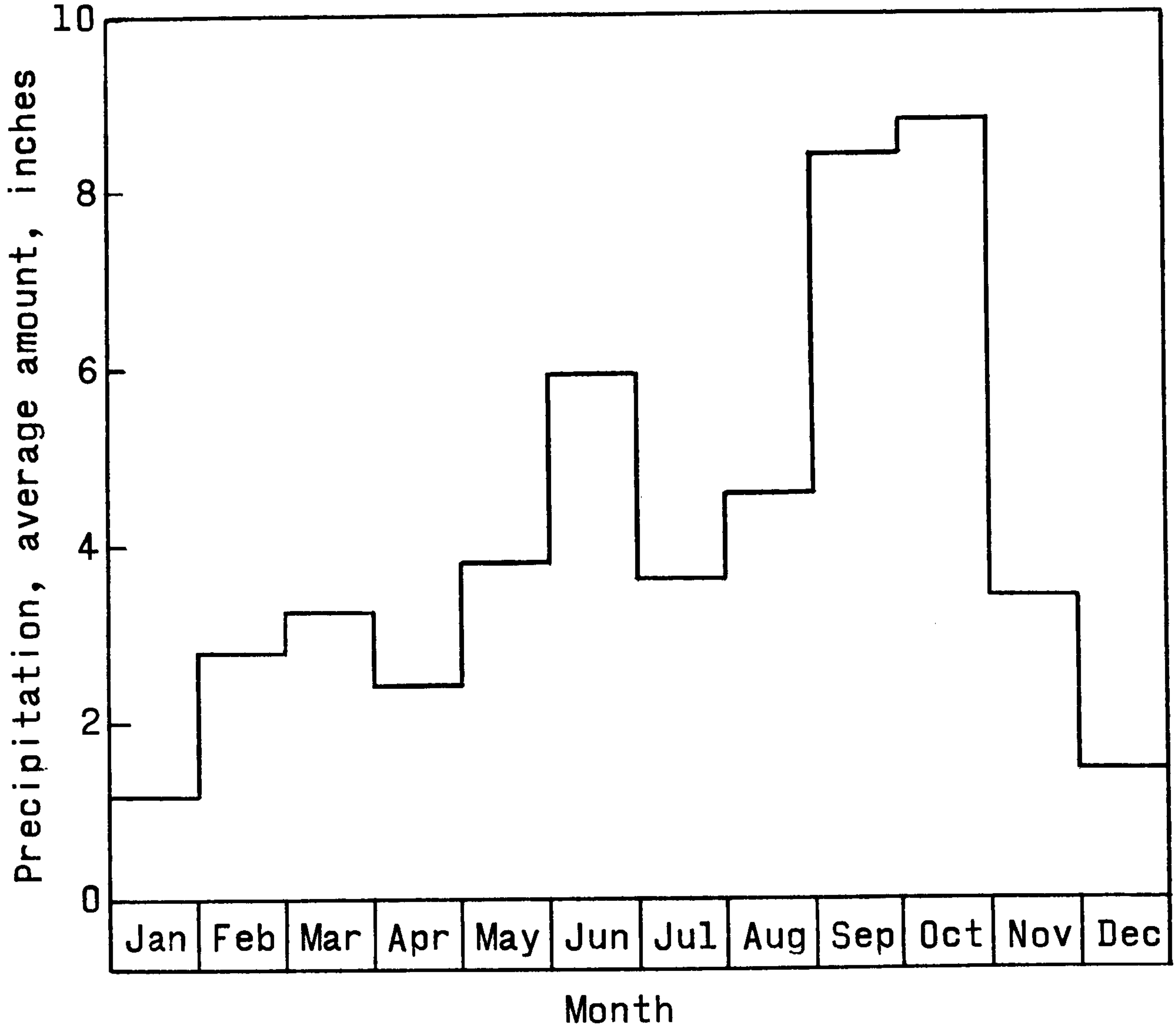


Figure 14.- Average monthly preceiptation, Patrick Air Force Base, Florida.

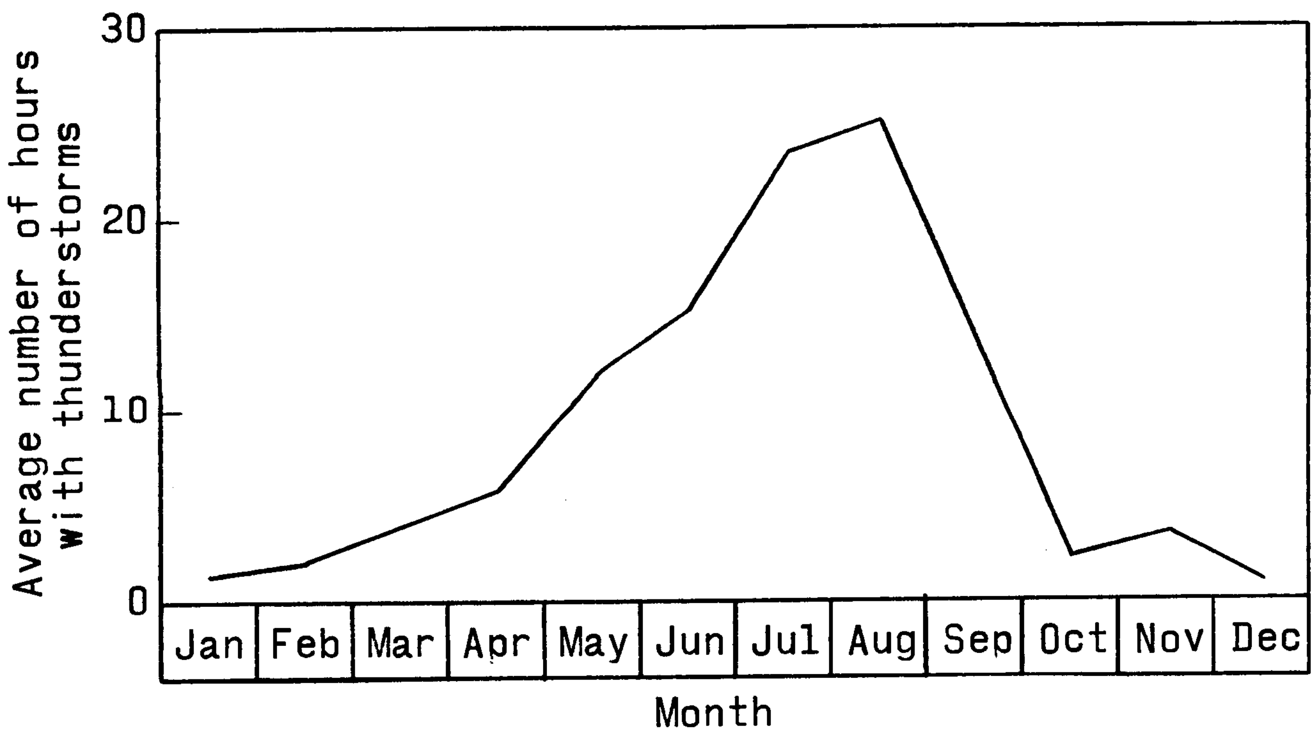


Figure 15.- Average number of hours with thunderstorms, Patrick Air Force Base, Florida.

[redacted]

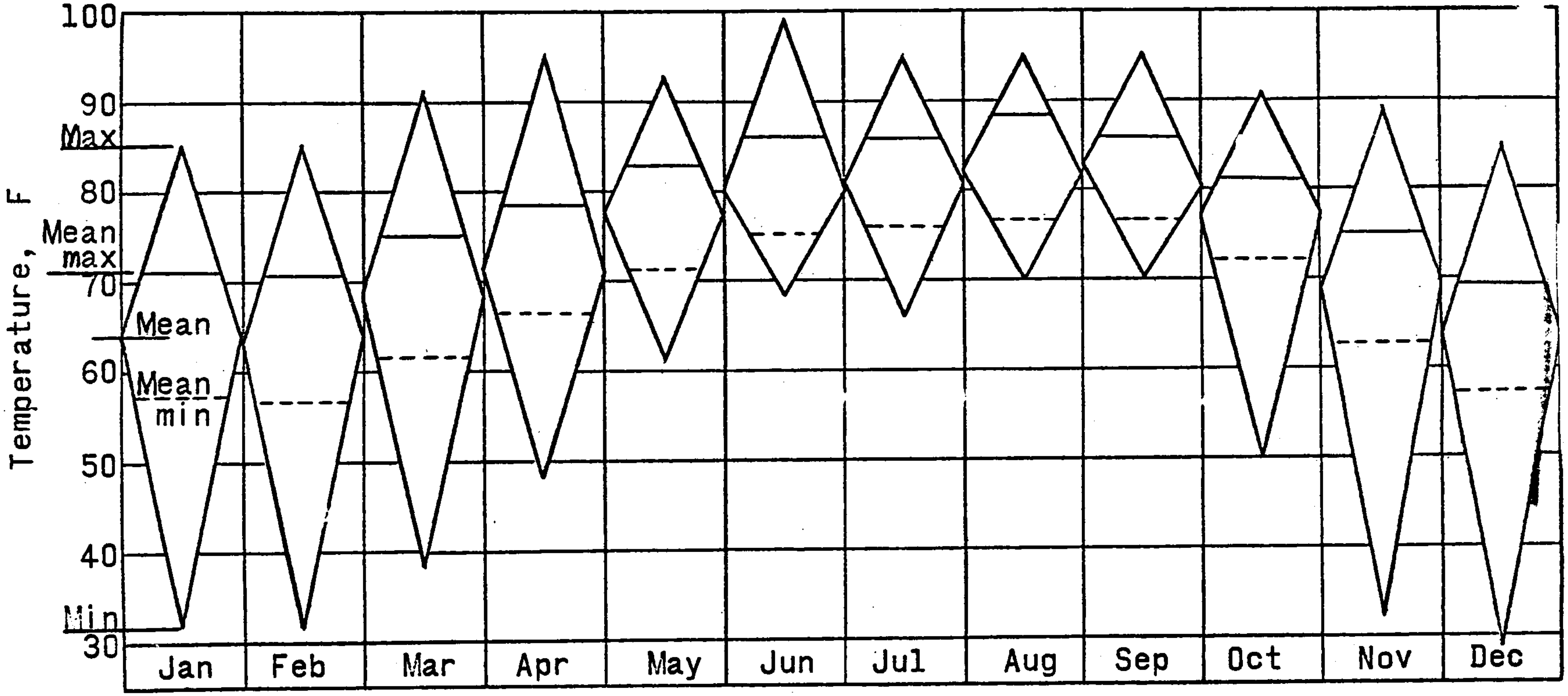


Figure 16.- Monthly temperature variations at Patrick Air Force Base, Florida.

[redacted]

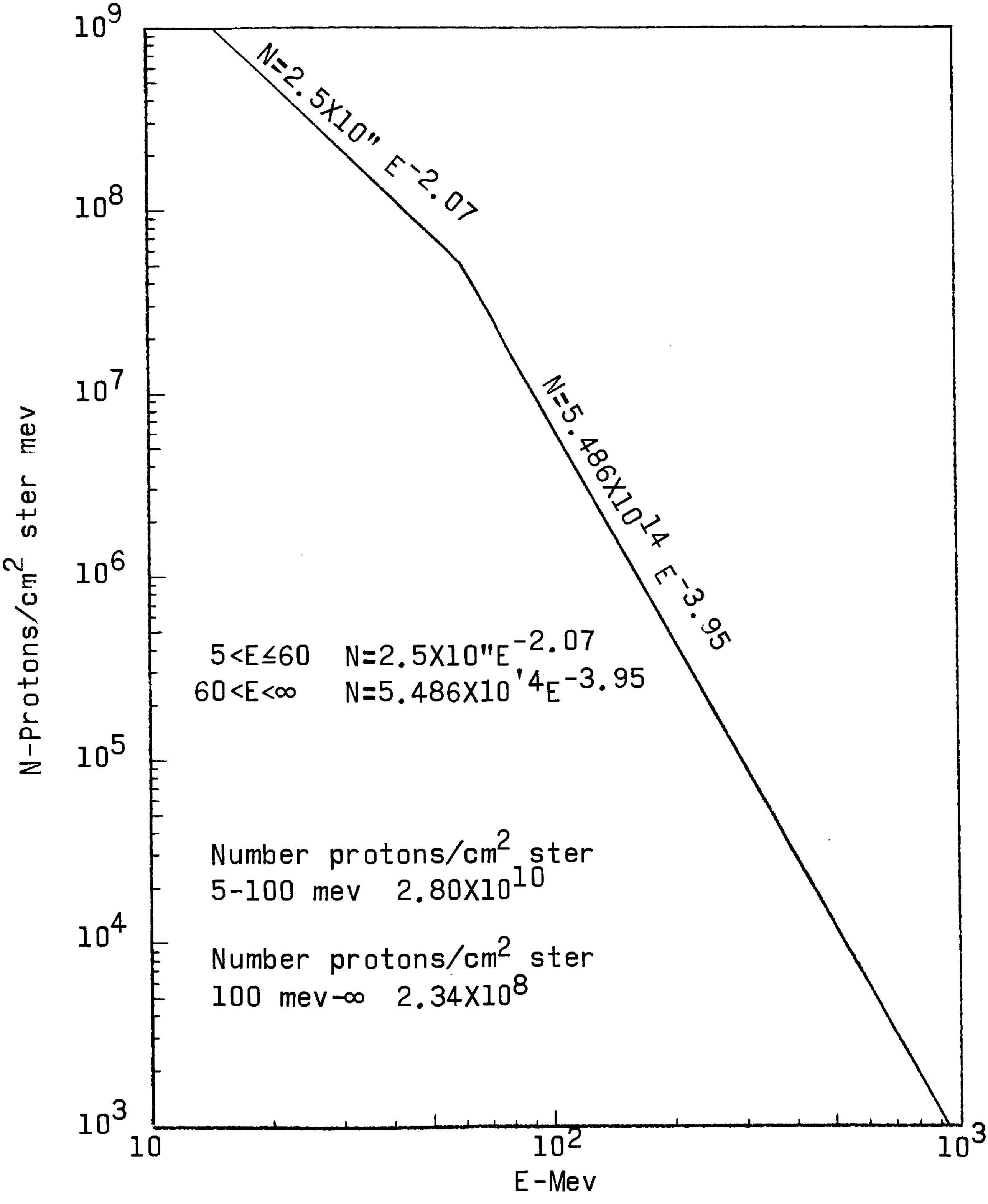


Figure 17.- Time integrated differential energy spectrum for May 10, 1959 event.

[redacted]

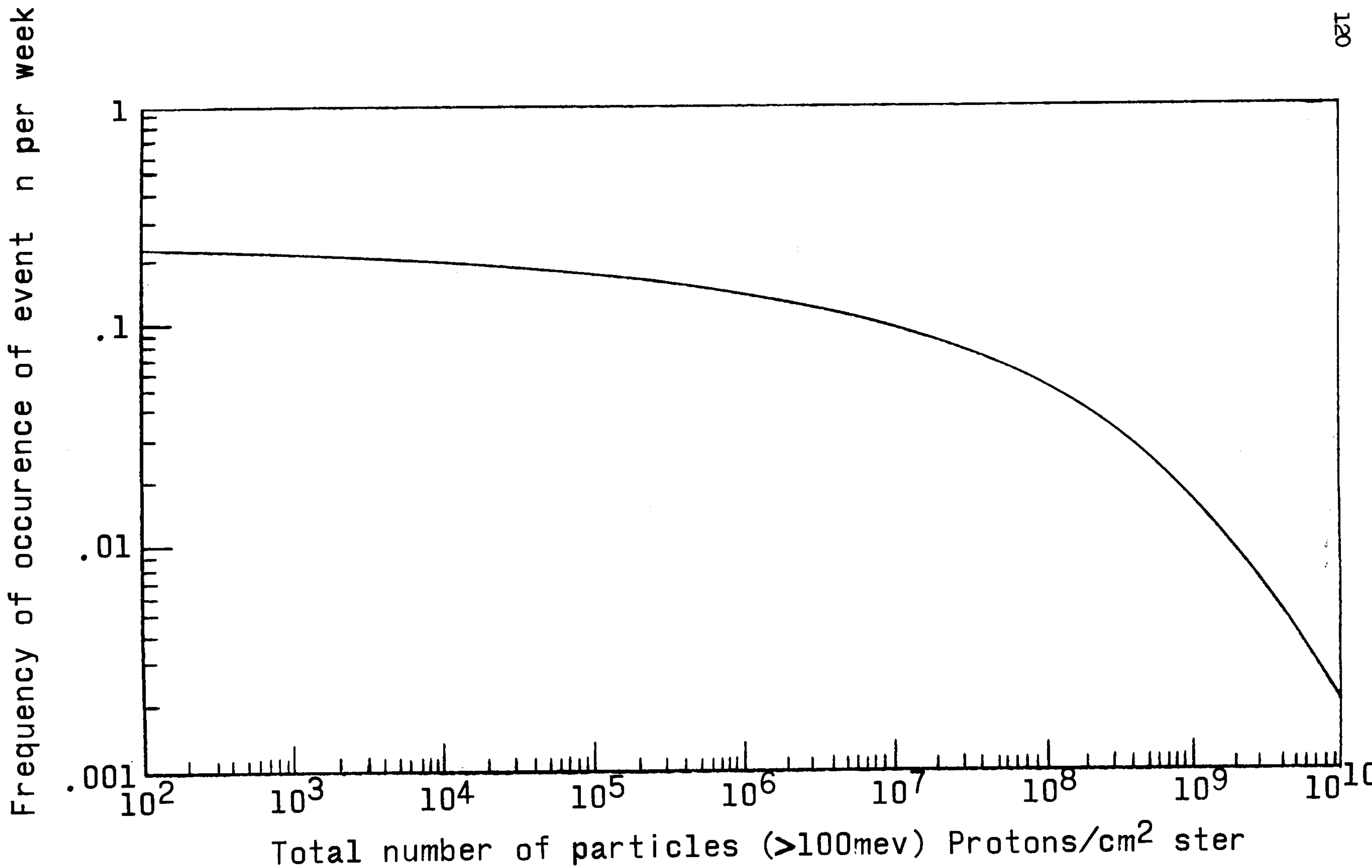


Figure 18.- Frequency distribution of solar proton events.

[redacted]

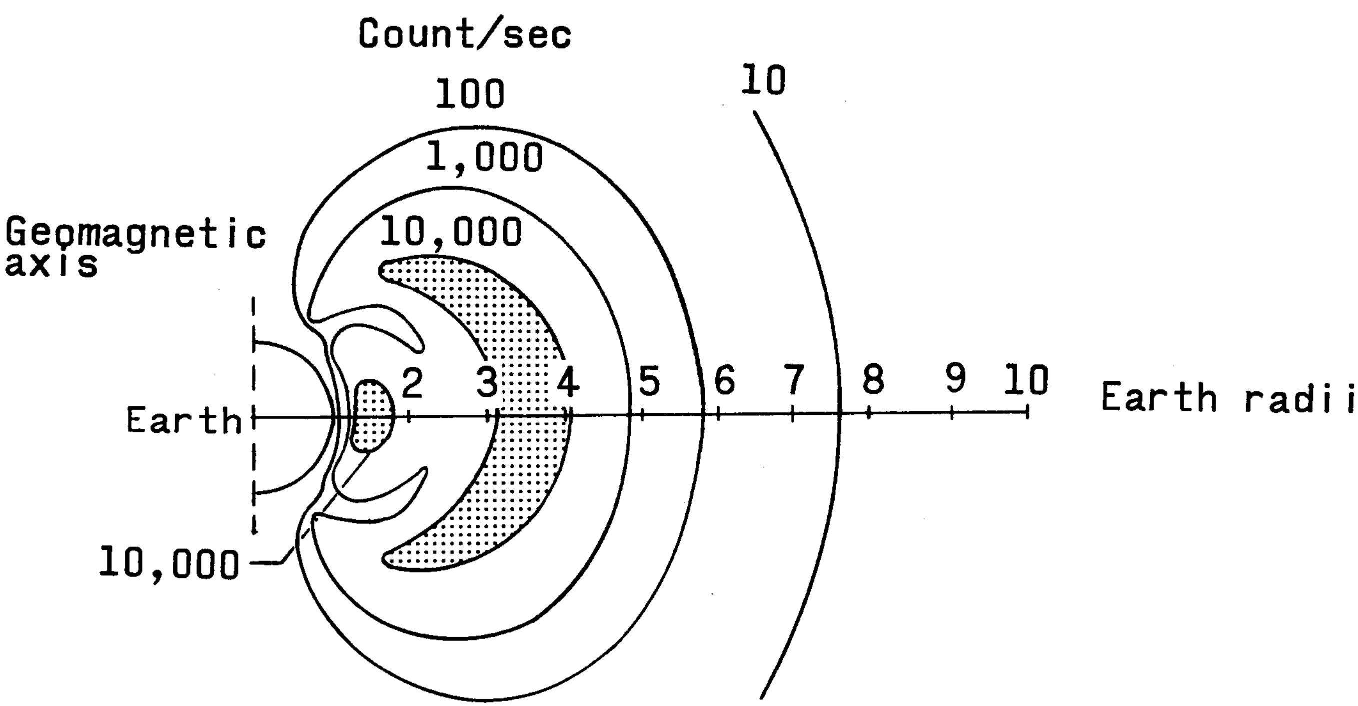


Figure 19.- Model of Van Allen radiation belts.

[redacted]

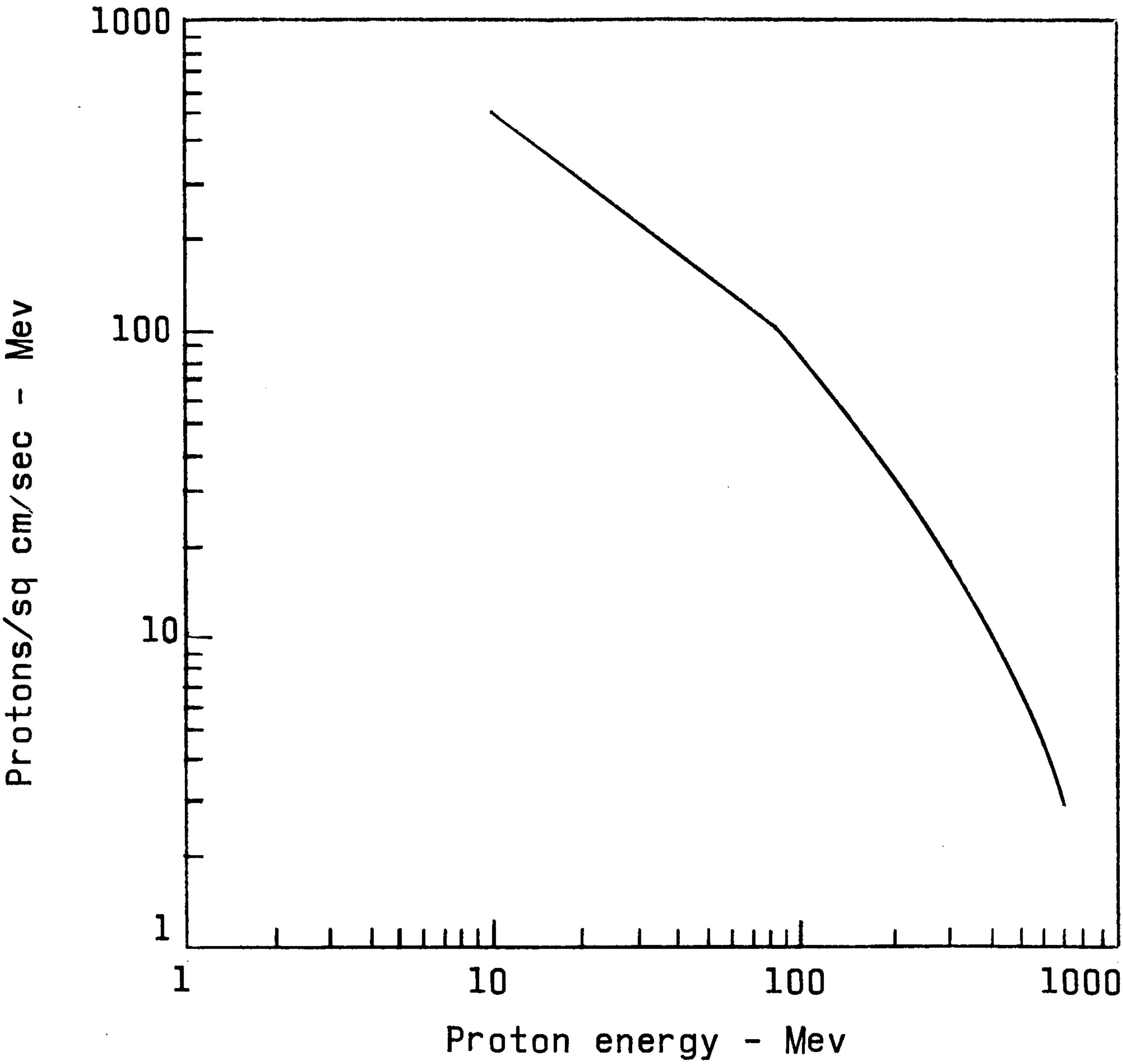


Figure 20.- Integral proton spectrum for the inner Van Allen belt at the geomagnetic equator.

[redacted]

Particles	Energy MEV	Particles/cm^2/sec
Electrons	>.04	10^8
Electrons	>2.4	10^5
Electrons	>5.0	10^2
Protons	>60	10^2
Protons	<30	No information

Figure 21.- Distribution of particles in the heart of the outer Van Allen belt.

[redacted]

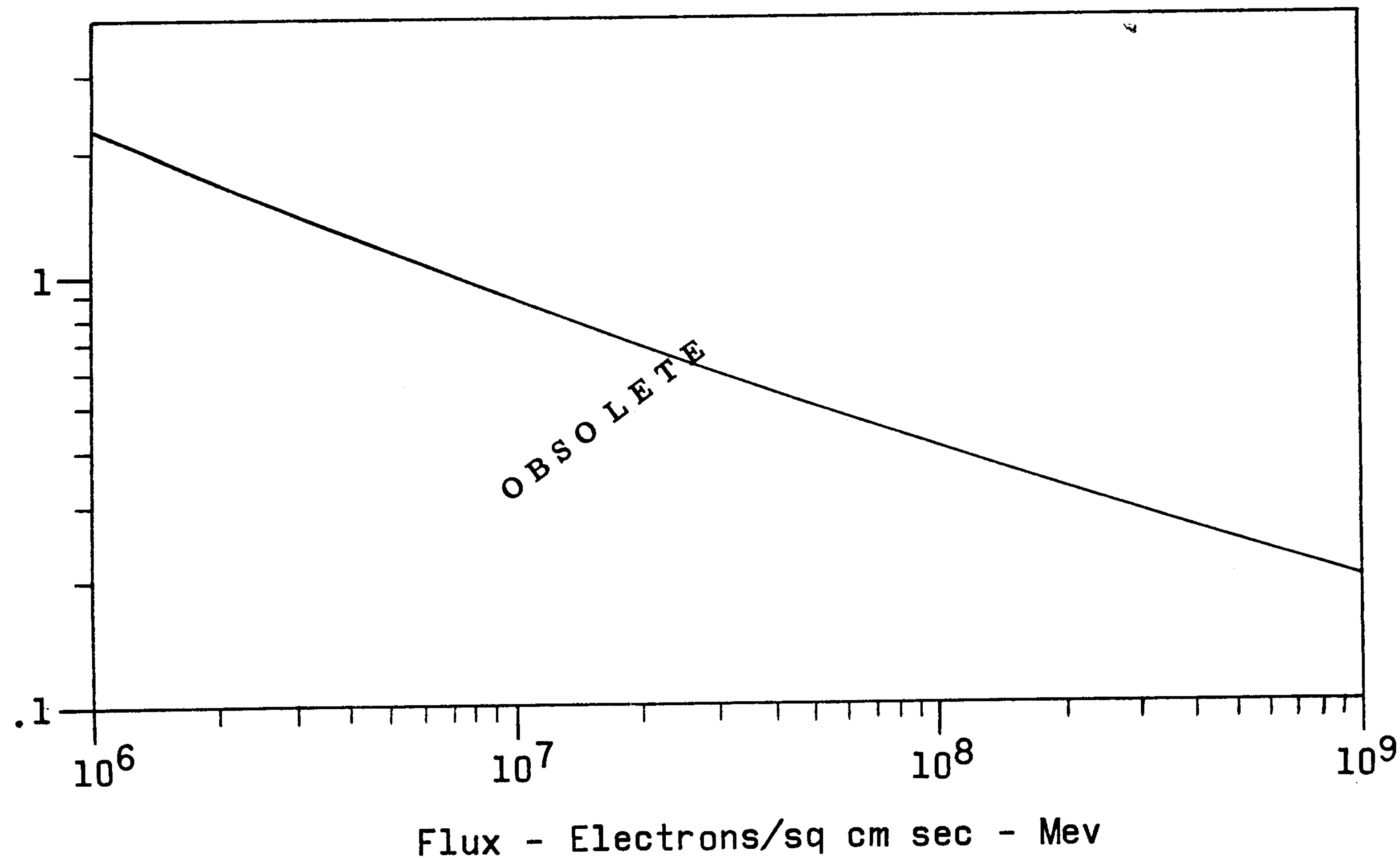


Figure 22.- Differential electron spectrum for the outer belt.

[redacted]

Visual magnitude	Mass Slugs	Mass grams	Diameter* microns	Diameter* inches	Daily accretion of earth	Velocity km/sec	Velocity ft/sec
0	$1.71x10^{-3}$	25.0	23,900	.940	$1.00x10^{6}$	28	91,900
1	$6.82x10^{-4}$	9.95	17,600	.693	$3.72x10^{6}$	28	91,900
2	$2.71x10^{-4}$	3.96	12,900	.508	$1.38x10^{7}$	28	91,900
3	$1.08x10^{-4}$	1.58	9,526	.374	$3.69x10^{7}$	28	91,900
4	$4.30x10^{-5}$	0.628	6,998	.275	$9.26x10^{7}$	28	91,900
5	$1.71x10^{-5}$	0.250	5,152	.203	$2.33x10^{8}$	28	91,900
6	$6.82x10^{-6}$	$9.95x10^{-2}$	3,791	.149	$5.84x10^{8}$	28	91,900
7	$2.71x10^{-6}$	$3.96x10^{-2}$	2,790	.110	$1.47x10^{9}$	28	91,900
8	$1.08x10^{-6}$	$1.58x10^{-2}$	2,051	$8.07x10^{-2}$	$3.69x10^{9}$	27	88,600
9	$4.30x10^{-7}$	$6.28x10^{-3}$	1,511	$5.96x10^{-2}$	$9.26x10^{9}$	26	85,300
10	$1.71x10^{-7}$	$2.50x10^{-3}$	1,113	$4.37x10^{-2}$	$2.33x10^{10}$	25	82,000
11	$6.82x10^{-8}$	$9.95x10^{-4}$	816	$3.21x10^{-2}$	$5.84x10^{10}$	24	78,700
12	$2.71x10^{-8}$	$3.96x10^{-4}$	603	$2.37x10^{-2}$	$1.47x10^{11}$	23	75,500
13	$1.08x10^{-8}$	$1.58x10^{-4}$	442	$1.74x10^{-2}$	$3.69x10^{11}$	22	72,200
14	$4.30x10^{-9}$	$6.28x10^{-5}$	325	$1.28x10^{-2}$	$9.26x10^{11}$	21	68,900
15	$1.71x10^{-9}$	$2.50x10^{-5}$	215	$9.40x10^{-3}$	$2.33x10^{12}$	20	65,600
16	$6.82x10^{-10}$	$9.95x10^{-6}$	176	$6.93x10^{-3}$	$5.84x10^{12}$	19	62,300
17	$2.71x10^{-10}$	$3.96x10^{-6}$	129	$5.08x10^{-3}$	$1.47x10^{13}$	18	59,100
18	$1.08x10^{-10}$	$1.58x10^{-6}$	95	$3.74x10^{-3}$	$3.69x10^{13}$	17	55,800
19	$4.30x10^{-11}$	$6.28.10^{-7}$	70	$2.75x10^{-3}$	$9.26x10^{13}$	16	52,500
20	$1.71x10^{-11}$	$2.50x10^{-7}$	51.5	$2.03x10^{-3}$	$2.33x10^{14}$	15	49,200
21	$6.82x10^{-12}$	$9.95x10^{-8}$	37.9	$1.49x10^{-3}$	$5.84x10^{14}$	15	49,200
22	$2.71x10^{-12}$	$3.96x10^{-8}$	27.9	$1.10x10^{-3}$	$1.47x10^{15}$	15	49,200
23	$1.08x10^{-12}$	$1.58x10^{-8}$	19.4	$8.12x10^{-4}$	$3.69x10^{15}$	15	49,200
24	$4.30x10^{-13}$	$6.28x10^{-9}$	12.2	$5.95x10^{-4}$	$9.26x10^{15}$	15	49,200
25	$1.71x10^{-13}$	$2.50x10^{-9}$	7.68	$4.37x10^{-4}$	$2.33x10^{16}$	15	49,200
26	$6.82x10^{-14}$	$9.95x10^{-10}$	4.86	$3.21x10^{-4}$	$5.84x10^{16}$	15	49,200
27	$2.71x10^{-14}$	$3.96x10^{-10}$	3.06	$2.37x10^{-4}$	$1.47x10^{17}$	15	49,200
28	$1.08x10^{-14}$	$1.58x10^{-10}$	1.95	$1.74x10^{-4}$	$3.69x10^{17}$	15	49,200
29	$4.30x10^{-15}$	$6.28x10^{-11}$	1.22	$1.28x10^{-4}$	$9.26.10^{17}$	15	49,200
30	$1.71x10^{-15}$	$2.50x10^{-11}$	.78	$9.38x10^{-5}$	$2.33x10^{18}$	15	49,200
31	$6.82x10^{-16}$	$9.95x10^{-12}$	.49	$6.87x10^{-5}$	$5.84x10^{18}$	15	49,200

*Diameter is based on ρ_m = 3.5 grams/cc

Figure 23.- Whipple's distribution for sporadic meteoroids.

[redacted]

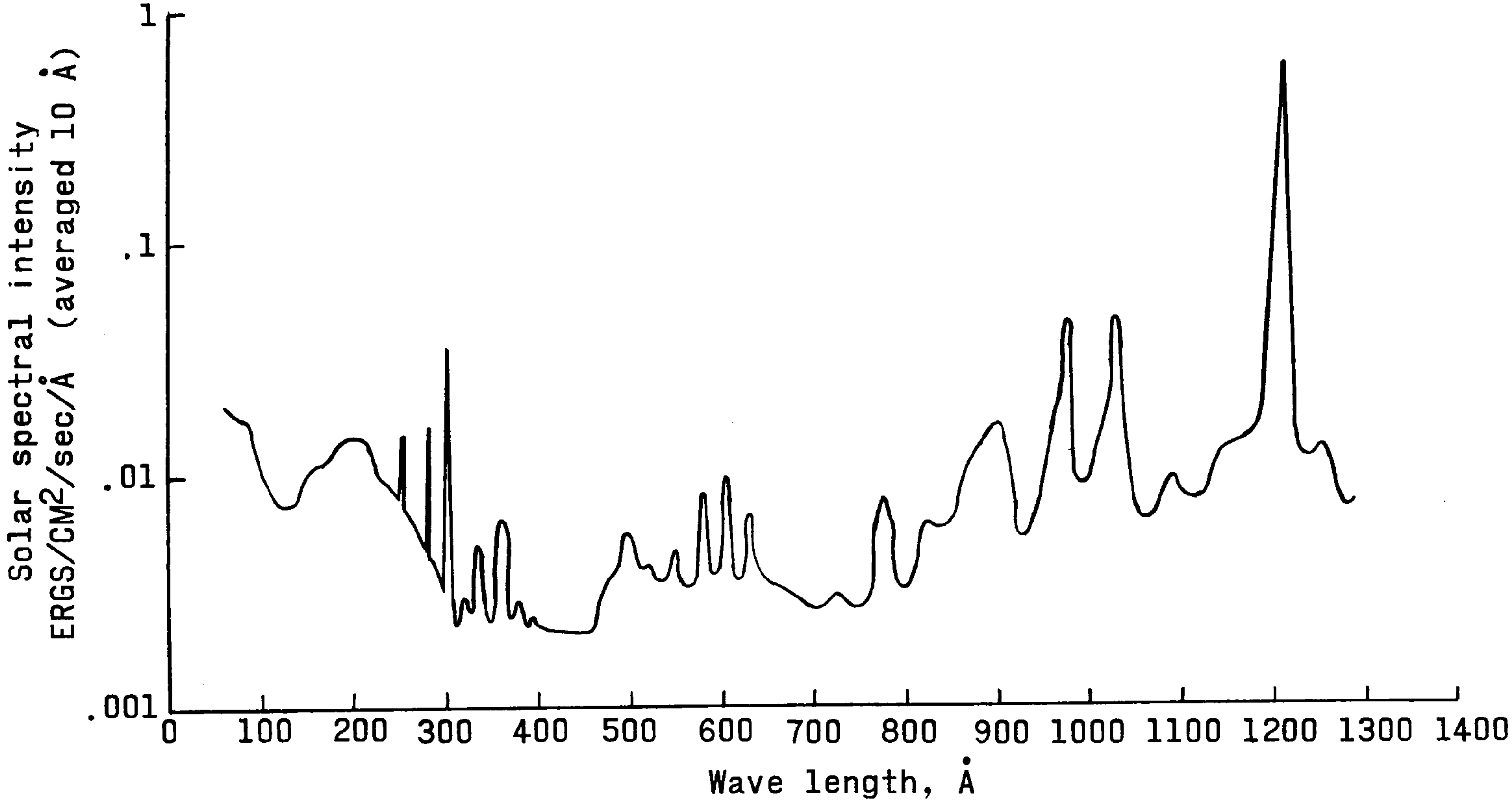


Figure 24.- Electromagnetic spectrum for solar radiation.

[redacted]

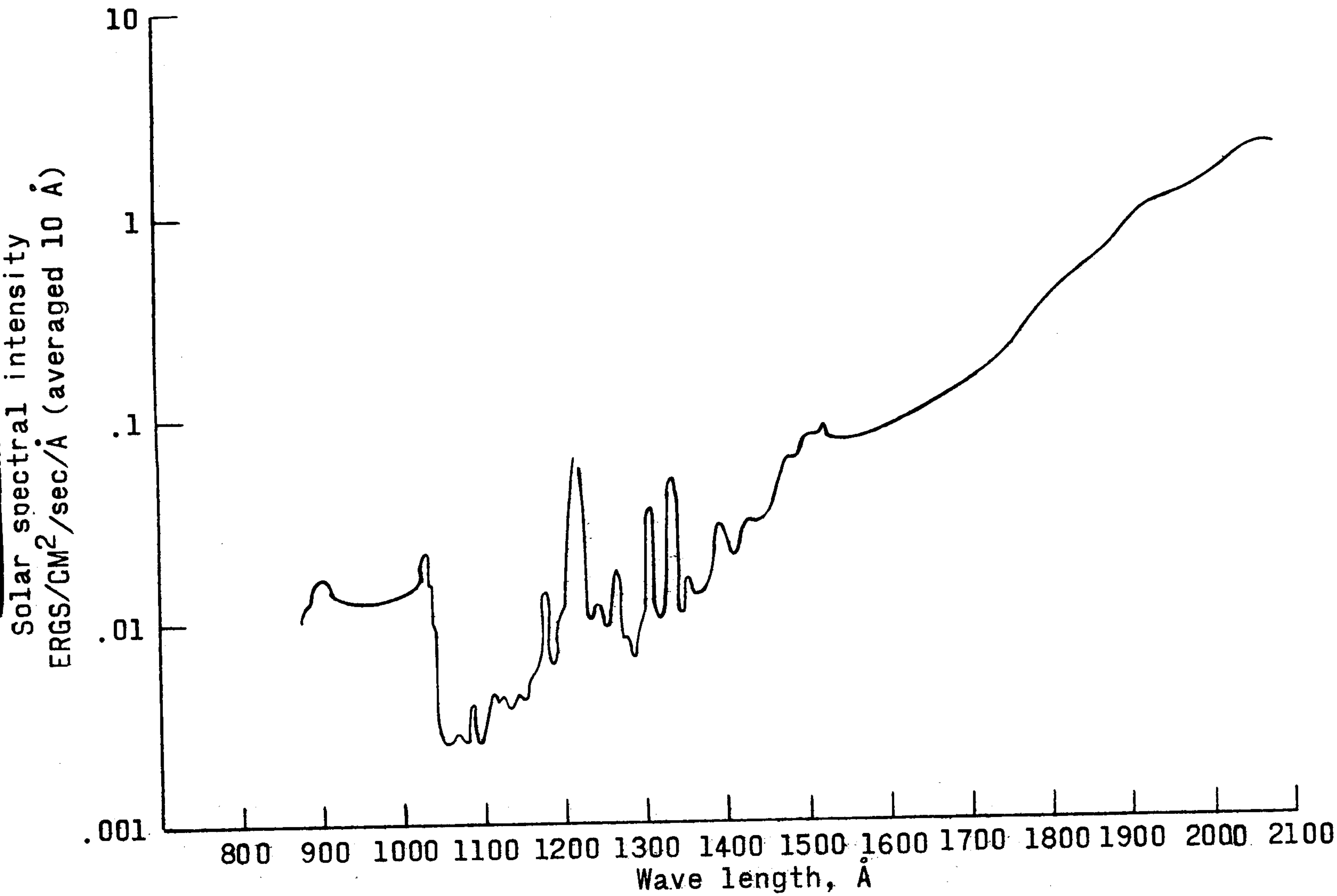


Figure 25.- Electromagnetic spectrum for solar radiation.

[redacted]

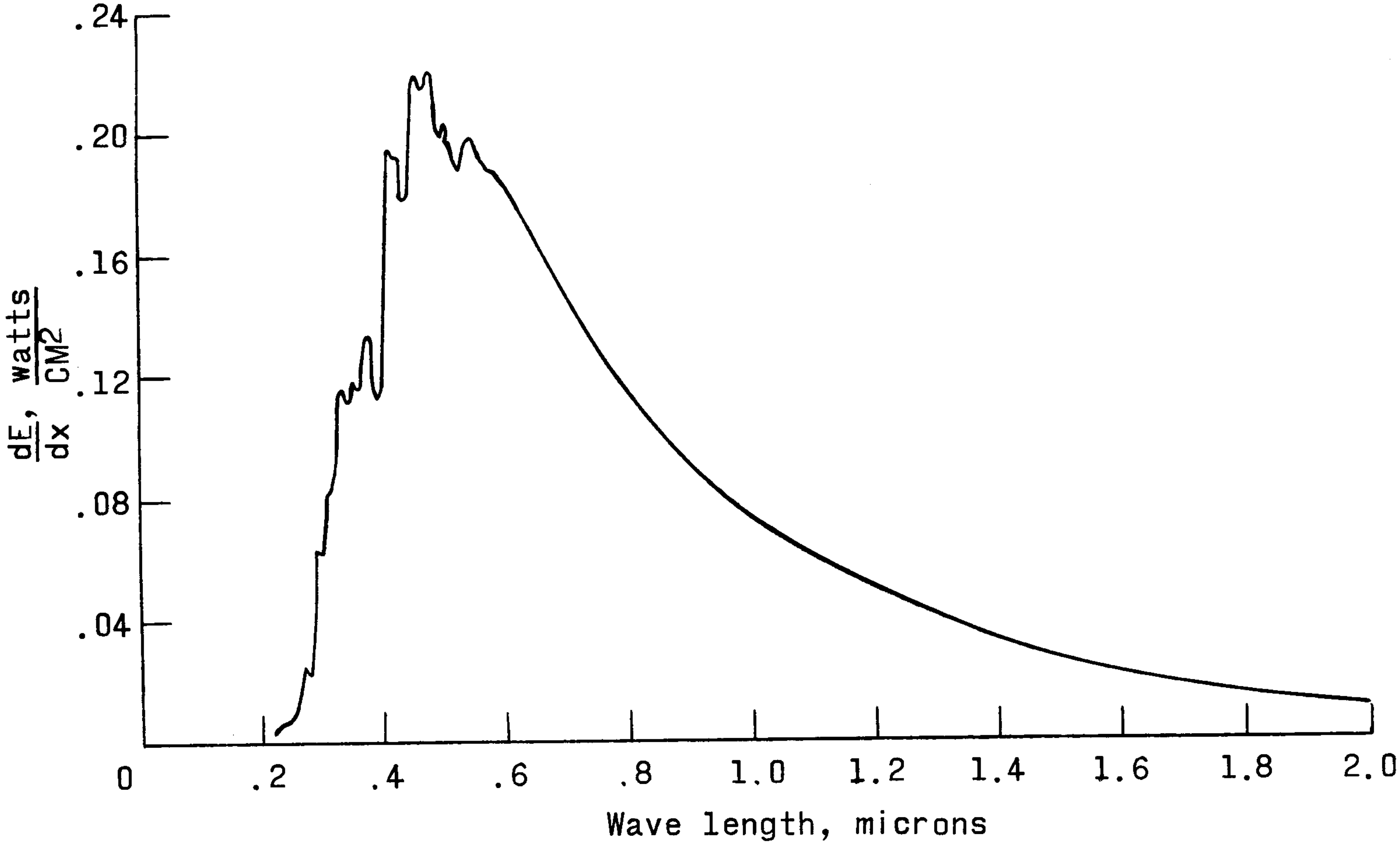


Figure 26.- Electromagnetic spectrum for solar radiation.

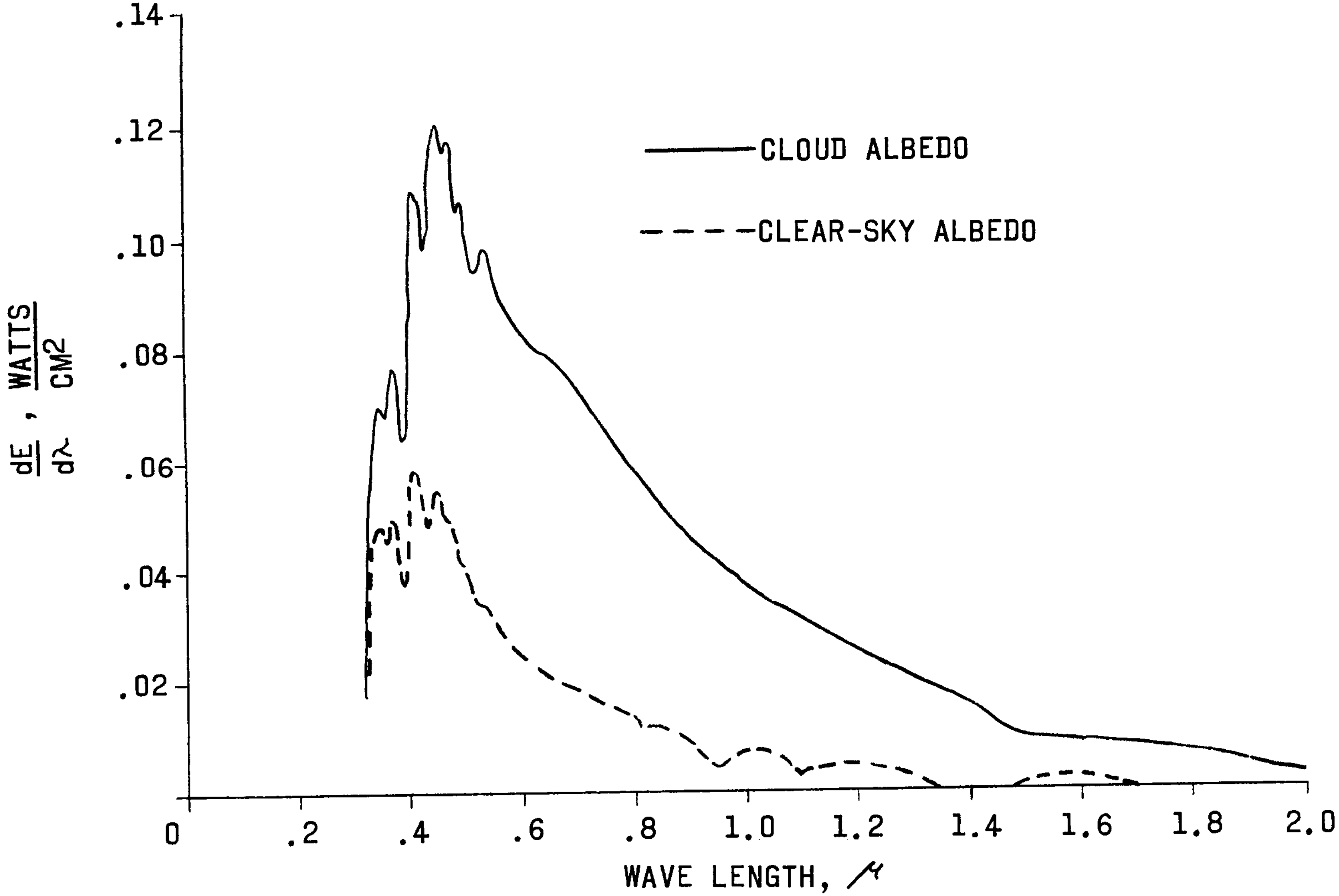


Figure 27.- Spectrum of the Earth's albedo.

A. Landing Site Topography

1. Craters

a. Average crater diameter is 80 miles.

b. The average crater wall height from the floor will be 14000 ft.

c. The average crater wall height above surrounding surface will be 7000 ft.

d. The slope of the crater walls will not exceed 30°.

e. The average slope of the crater floor will be 5° or less.

f. Protuberances or depressions greater than 10 cm may be avoided by landing area selection and/or lateral maneuvering up to 500 ft.

2. Maria

a. The highest mountain ranges and isolated peaks will not exceed 30,000 ft.

b. The average slope of the mountain ranges and isolated peaks will not exceed 30°.

c. Rills or cracks in the maria surface crater may be avoided by landing area selection and/or lateral maneuvering up to 500 ft.

d. The general maria will have a slope of less than 3°.

B. Soils Mechanics of Maria Surfaces and Crater Floors

a. The structure consists of a layer of loose material on the immediate surface whose average thickness does not exceed 10 cm and will have a particle size gradient from particles .3 mm in diameter on the immediate surface to 2 mm at the termination of the layer. The density is 3.3 grams/cc for individual particles.

Figure 28.- Model lunar surface.

b. The layer of loose material rests on the continuous layer of rock froth, whose average thickness does not exceed 30 cm.

c. The rock froth merges with the semi-continuous solid rock layer.

d. The average bearing strength of the dust layer is 12 psi, the rock froth 200 psi and the solid layer rock 400 psi.

C. Thermal Characteristics

a. The thermal conductivity of the dust layer is in the order of 10^{-6} cgs units. The product of thermal conductivity, density and heat capacity will be 10^{-4} cgs units. The $K\rho c$ product for the rock froth layer will be 10^{-3} cgs units and the solid rock layer 10^{-2} cgs units.

Figure 28.- Concluded.

[redacted]

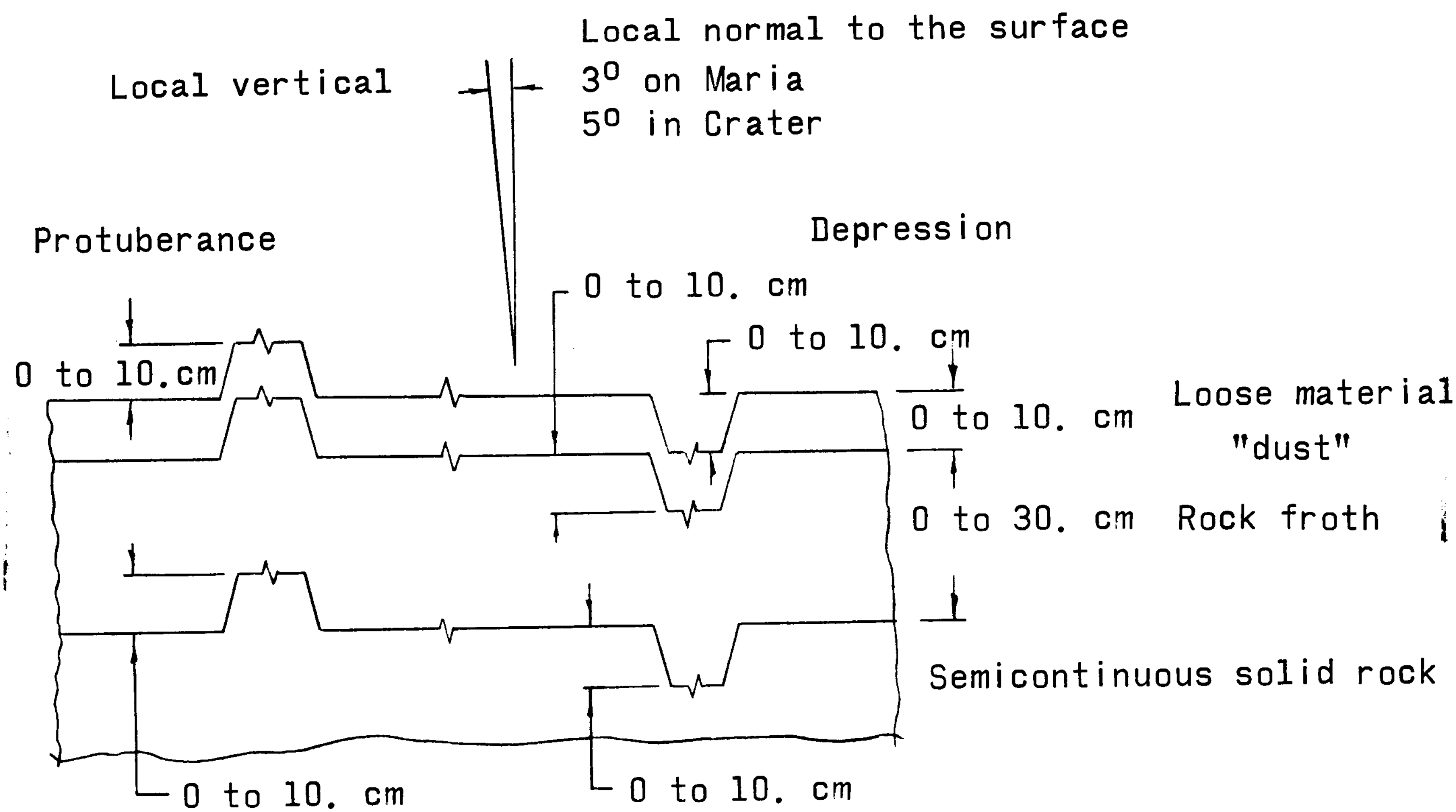


Figure 29.- Cross section of model lunar surfaces.

[redacted]

Equivalent pressure	Equivalent density	Composition
10^{-6} dynes/cm^2	10^{-16} grams/cm^3	Hydrogen atoms

Figure 30.- Interplanetary atmosphere.

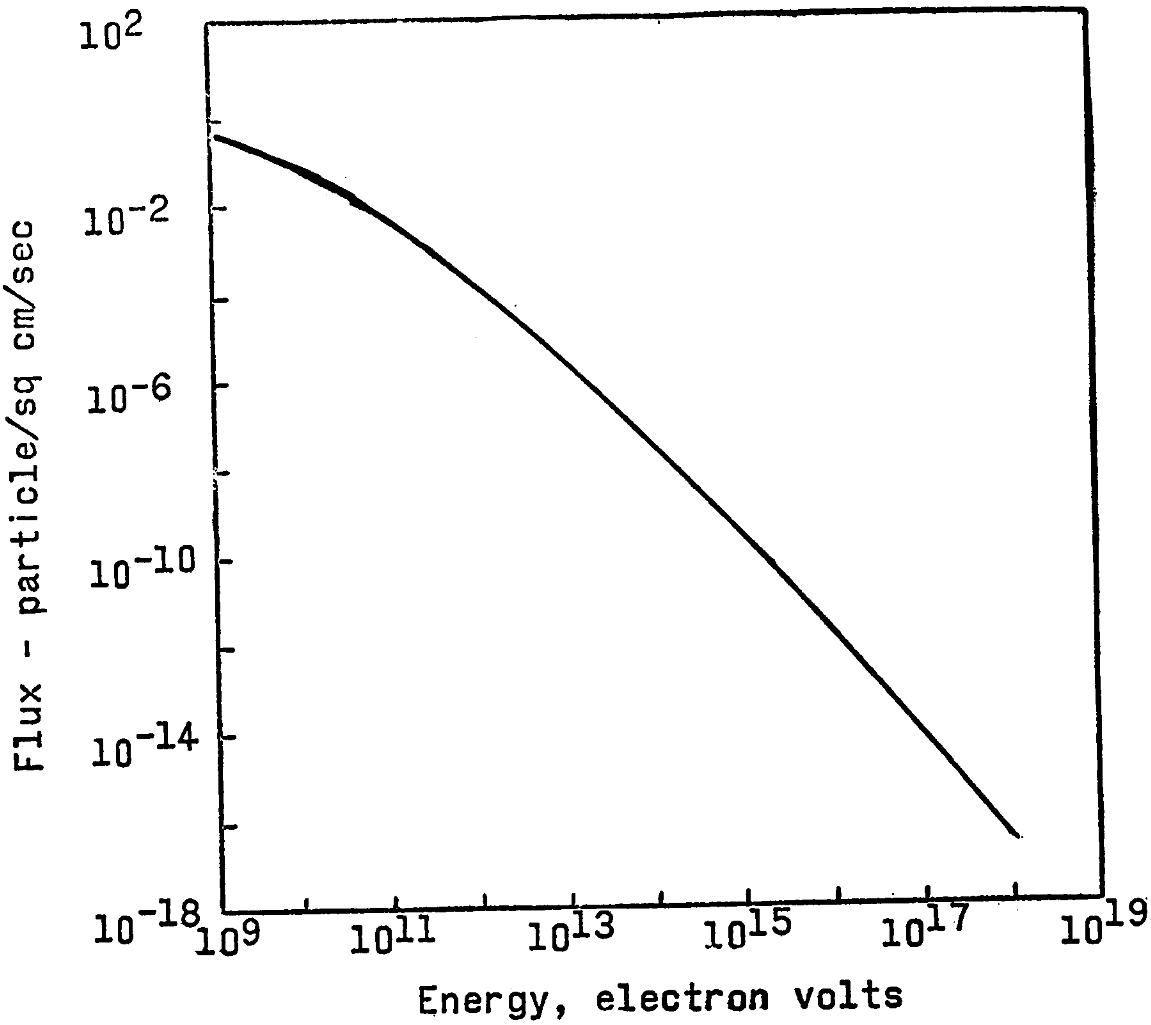


Figure 31.- Galactic cosmic ray flux.

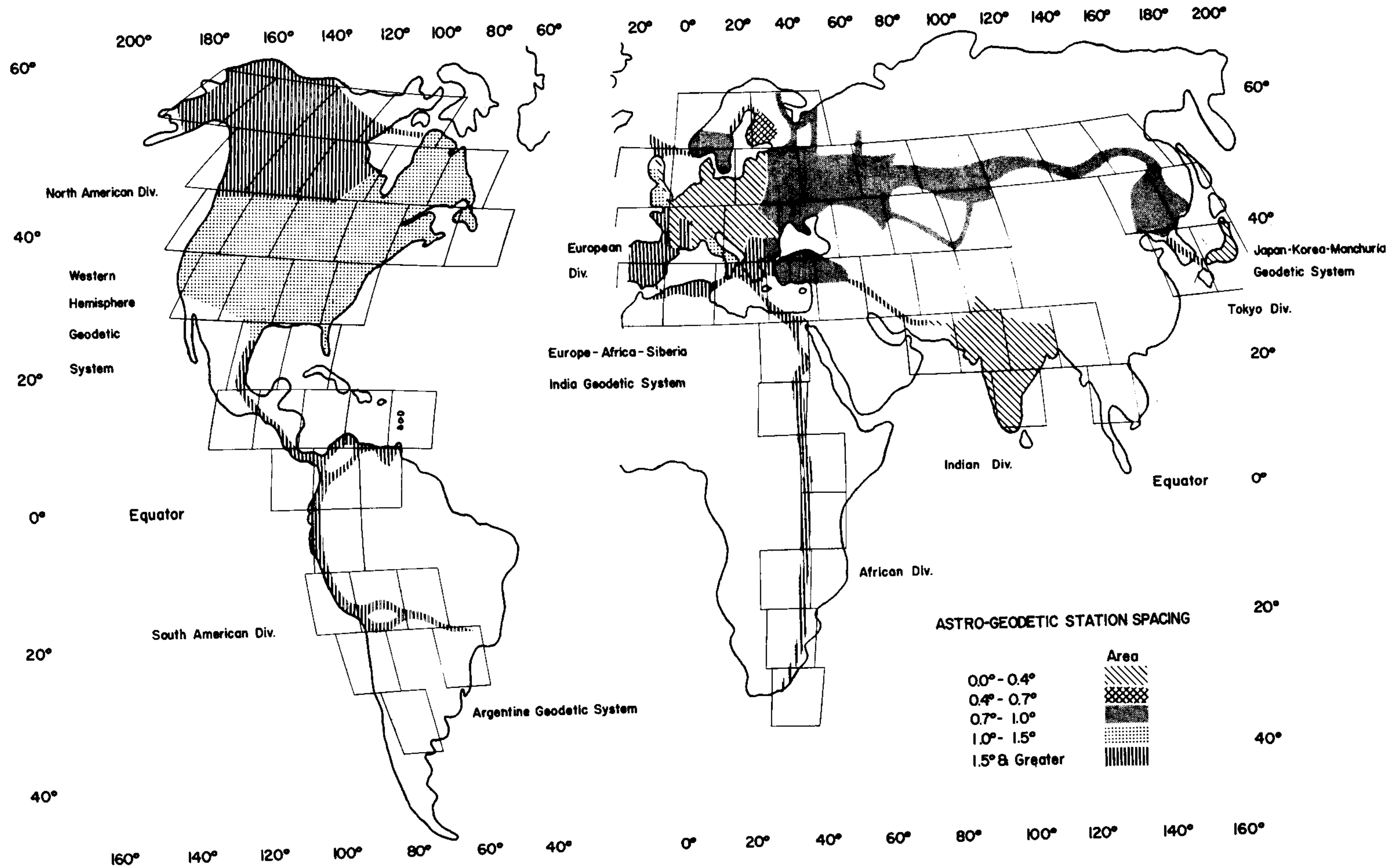


Figure 32.- Astro-geodetic geoid data station spacing and distribution.

Systems	Stations	Correction Δu Meters	Δv Meters	Δw Meters
Western Hemisphere Geodetic System	NAD	-23	+142	+196
	SAD	-303	+98	-315
	SAO SP59	+4	+299	+15
	Vanguard	-12	+235	+120
	σ	±26	±22	±22
Europe-Africa-Siberia-India Geodetic System	ED	-57	-37	-96
	Indian	+200	+782	+271
	Arc	-109	-70	-289
	SAO SP59	-150	-2	-33
	σ	±23	±29	±23
Japan-Korea-Manchuria Geodetic System	Tokyo	-89	+551	+710
	SAO SP59	-29	-209	-147
	σ	±40	±53	±40
Australia Geodetic System	Sidney	+198	+262	-21
	SAO SP59	+149	-83	+116
	σ (Estimated)	±75	±90	±35
Argentina Geodetic System	SAO SP59	-81	+131	+105
	σ (Estimated)	±180	±160	±160

Figure 33.- Geodetic station location correction data. (Reference 11).

Planet	M_s/M_p	ω	f	GM
Sun	1.	3.0050435×10^{-6}	0	$1.32715445 \times 10^{11}$
Mercury	6,120,000.		0	
Venus	406,645.	Synchronous	0	3.247695×10^{5}
Earth	332,488.		1/298.30	3.986032×10^{5}
Mars	3,088,000.	7.0882232×10^{-5}	1/191.8	4.297780×10^{4}
Jupiter	1,047.39	1.7734082×10^{-4}	1/15.2	1.267106×10^{8}
Saturn	3,500.	1.7055335×10^{-4}	1/10.2	
Uranus	22,869.	1.6135556×10^{-4}	1/14.	
Neptune	18,889.	1.1140400×10^{-4}	1/58.5	
Pluto	400,000.			

$$G = (6.668 \pm .0005) \times 10^{-8} \frac{c_m^3}{sec^2 gram}$$

$$M_e = (5.977 \pm .004) \times 10^{27} \text{ grams}$$

$$T = 86164.09054 \text{ seconds}$$

$$A_u = 1.49599 \times 10^{11} \text{ meters}$$

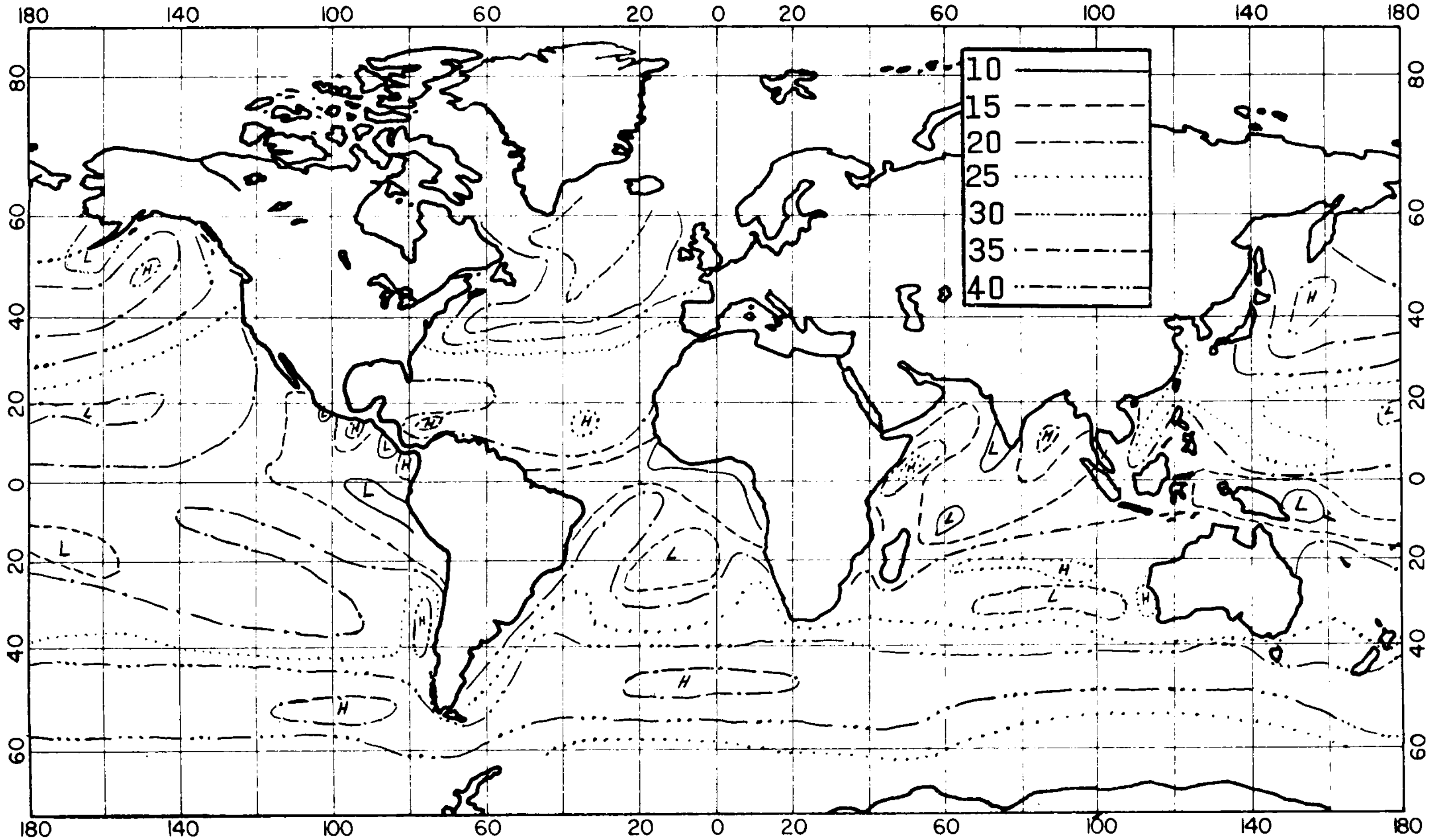
$$\frac{M_e}{M_m} = 81.375$$

Figure 34.- Sun, moon, and planetary constants.

[redacted]

[redacted]

[redacted]



(a) January.

Figure 35.- Wind speed (knots) exceeded 10 percent of the time.

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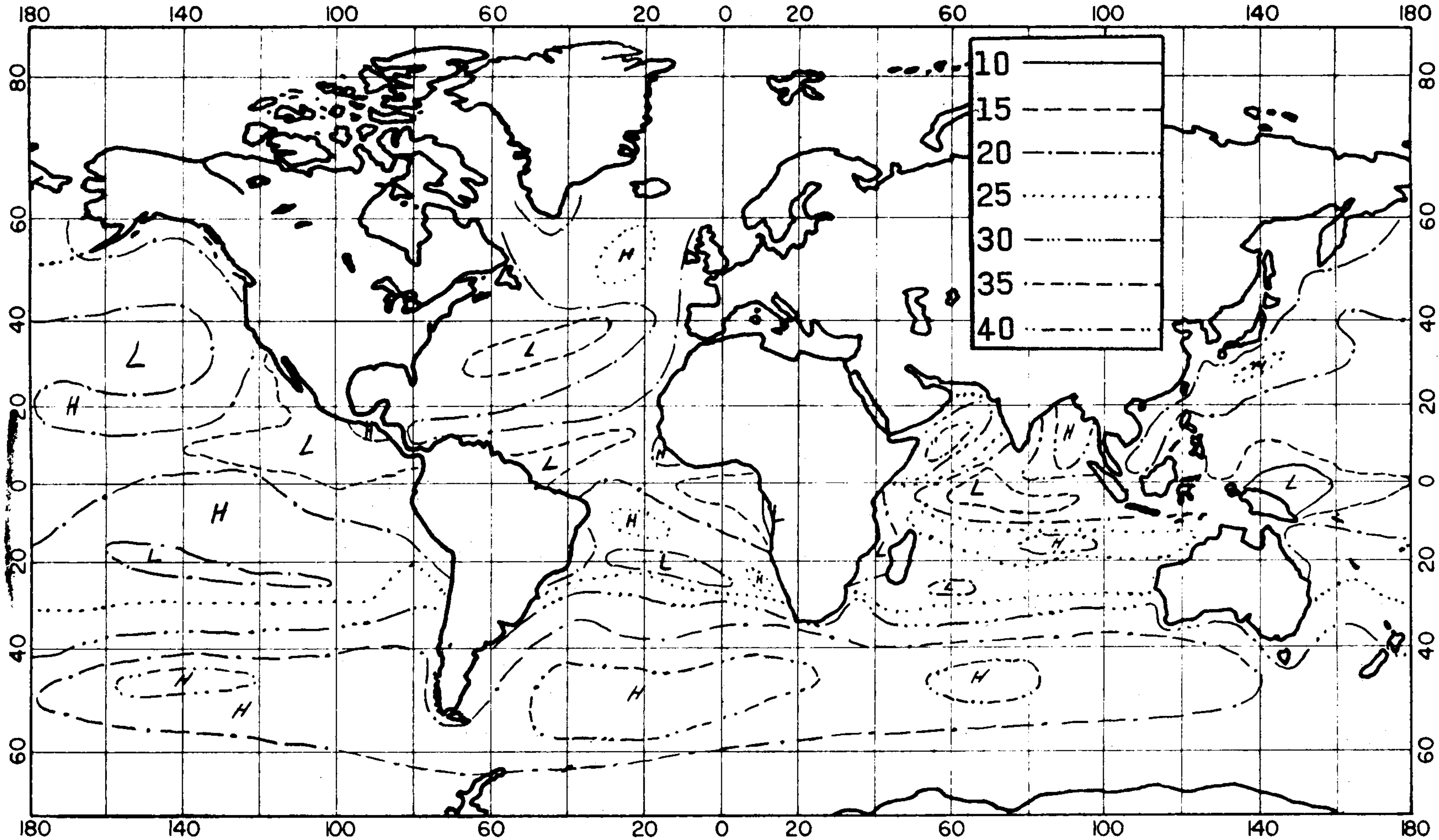
[redacted]



(b) July.

Figure 35.- Concluded.

[redacted]

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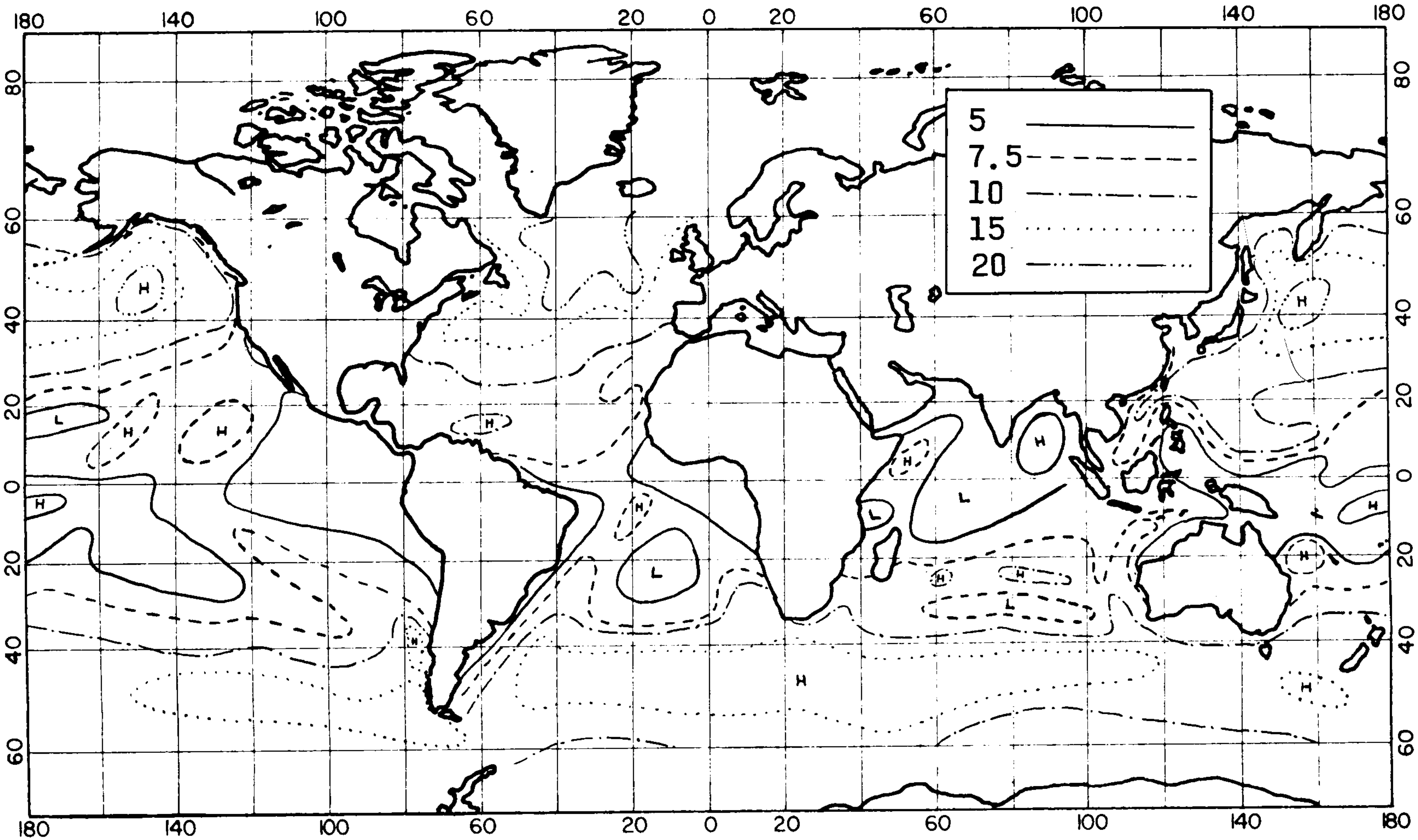
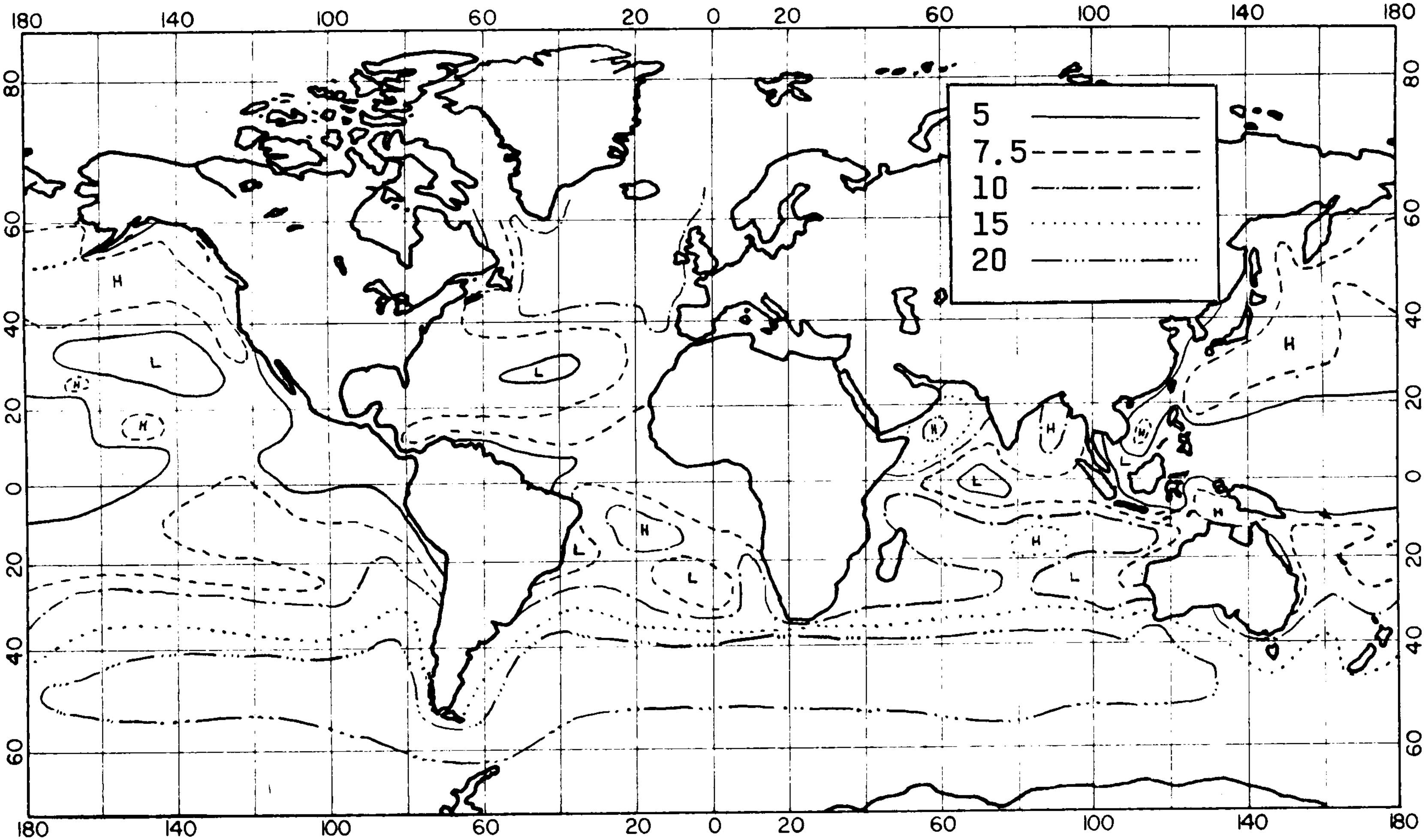


(a) January.

Figure 36.- Wave height (feet) exceeded 10 percent of the time.

[redacted]

[redacted]



[redacted]

(b) July.

Figure 36.- Concluded.

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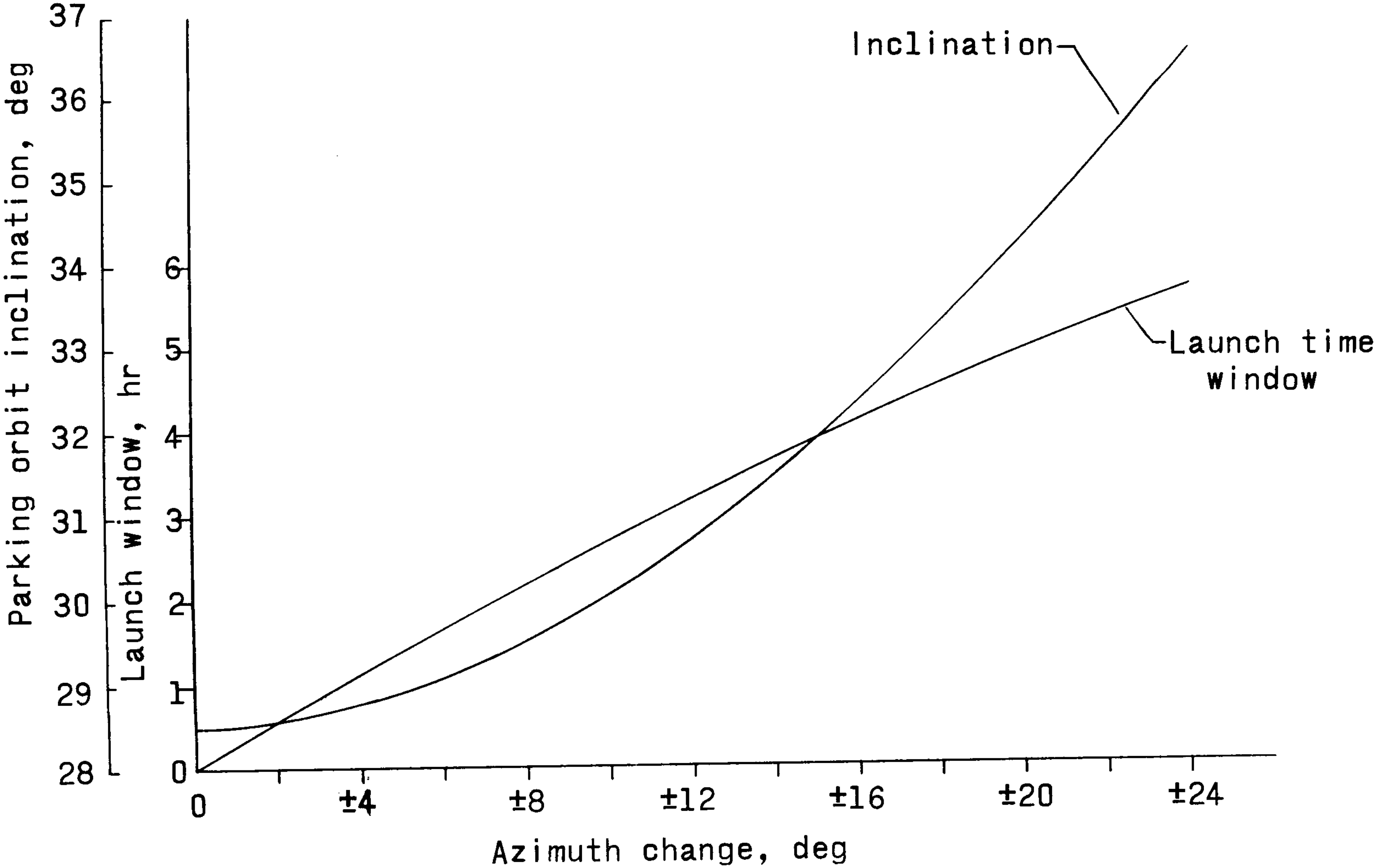


Figure 37.- Launch time window for variation in the translunar trajectory.

[redacted]

[redacted]

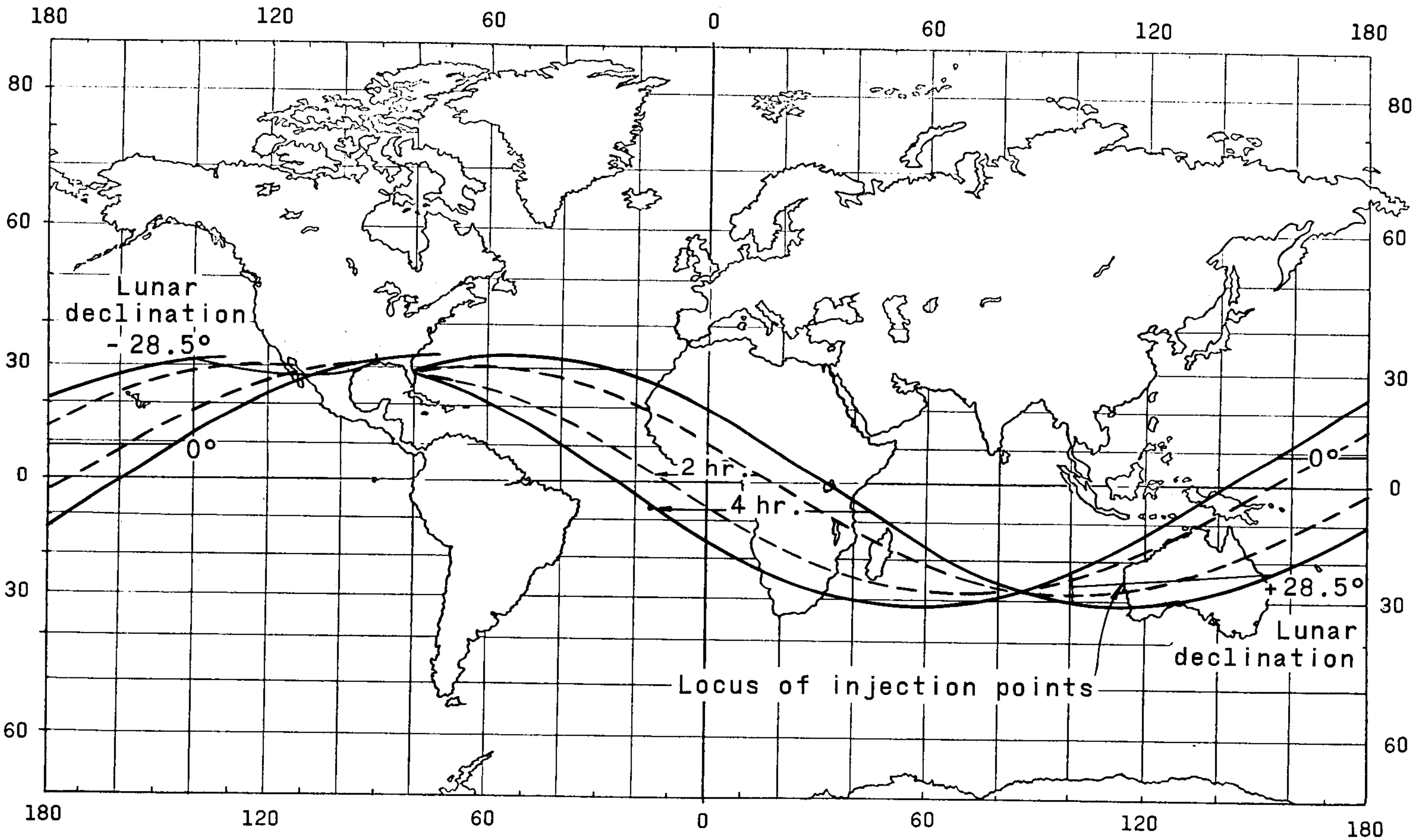
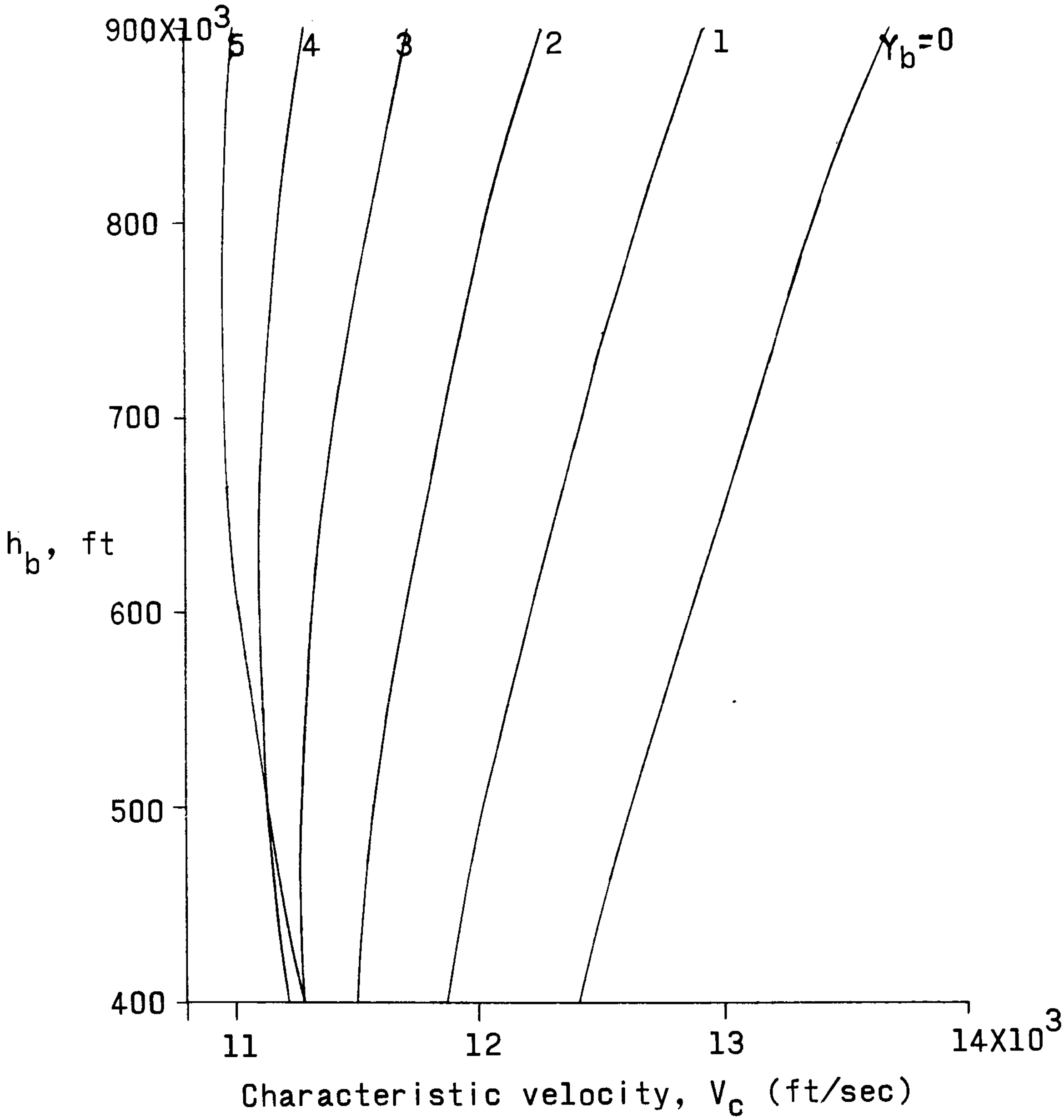


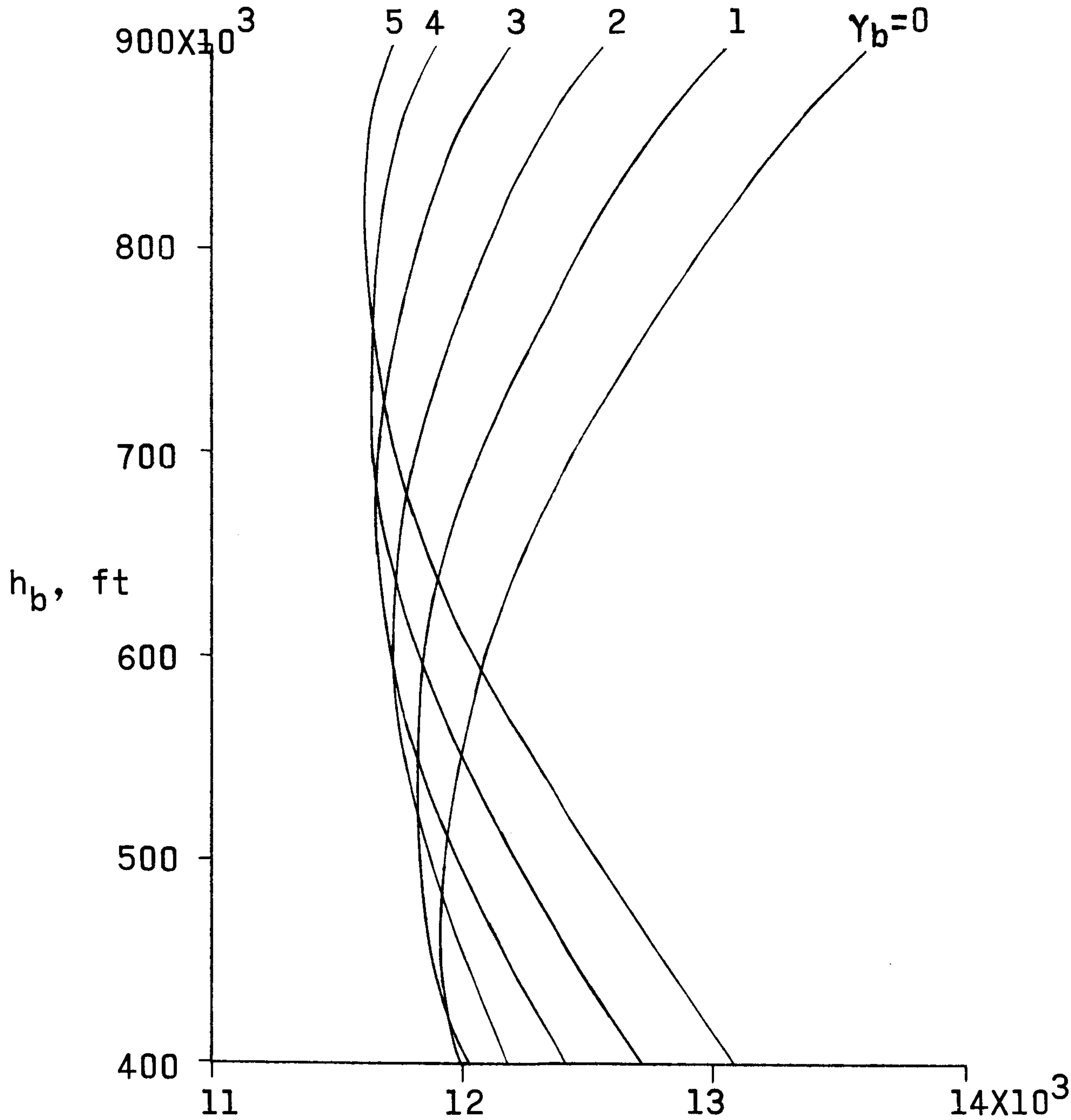
Figure 38.- Parking orbit boundaries for launch time window.

[redacted]



(a) T/W = 0.5, I_{sp} = 420

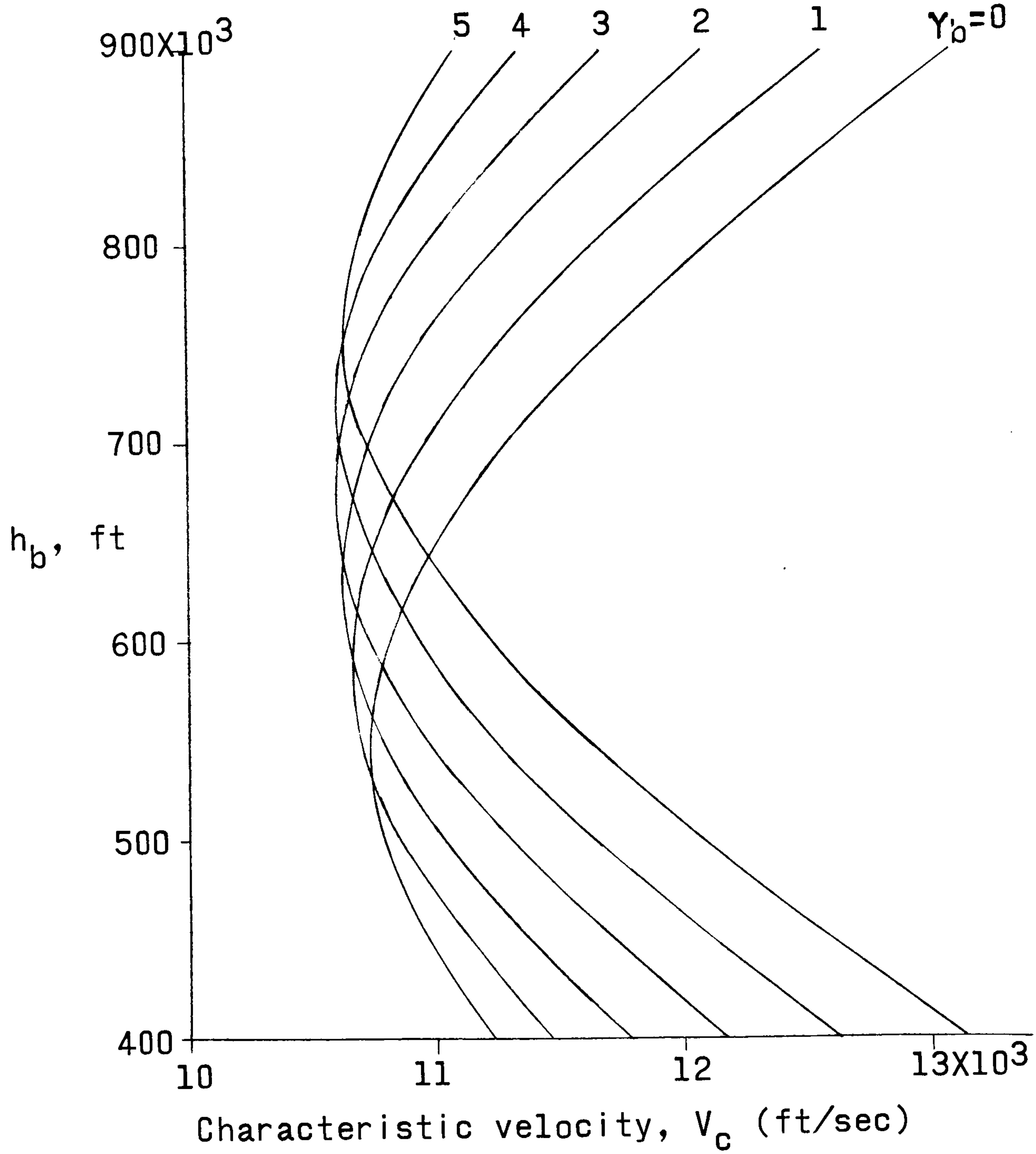
Figure 39.- Optimum booster performance to escape from a 600,000 ft parking orbit.



(b) $T/W = 1.0$, $I_{sp} = 420$

Figure 39.- Continued.

[redacted]



(c) $T/W = 1.50$, $I_{sp} = 420$

Figure 39.- Concluded.

[redacted]

[redacted]

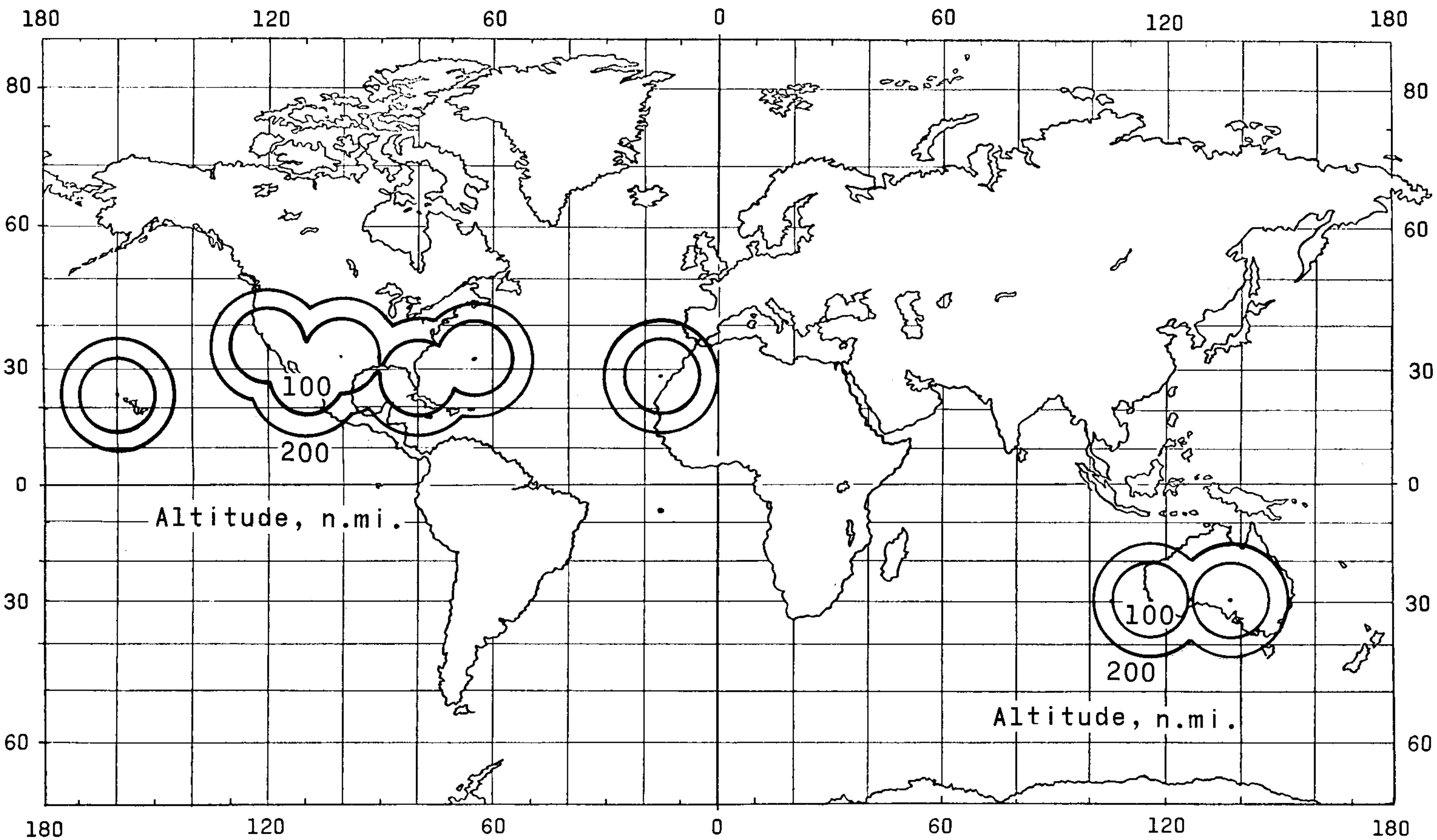


Figure 40.- Coverage of Mercury tracking network for 5° elevation angle.

[redacted]

[redacted]

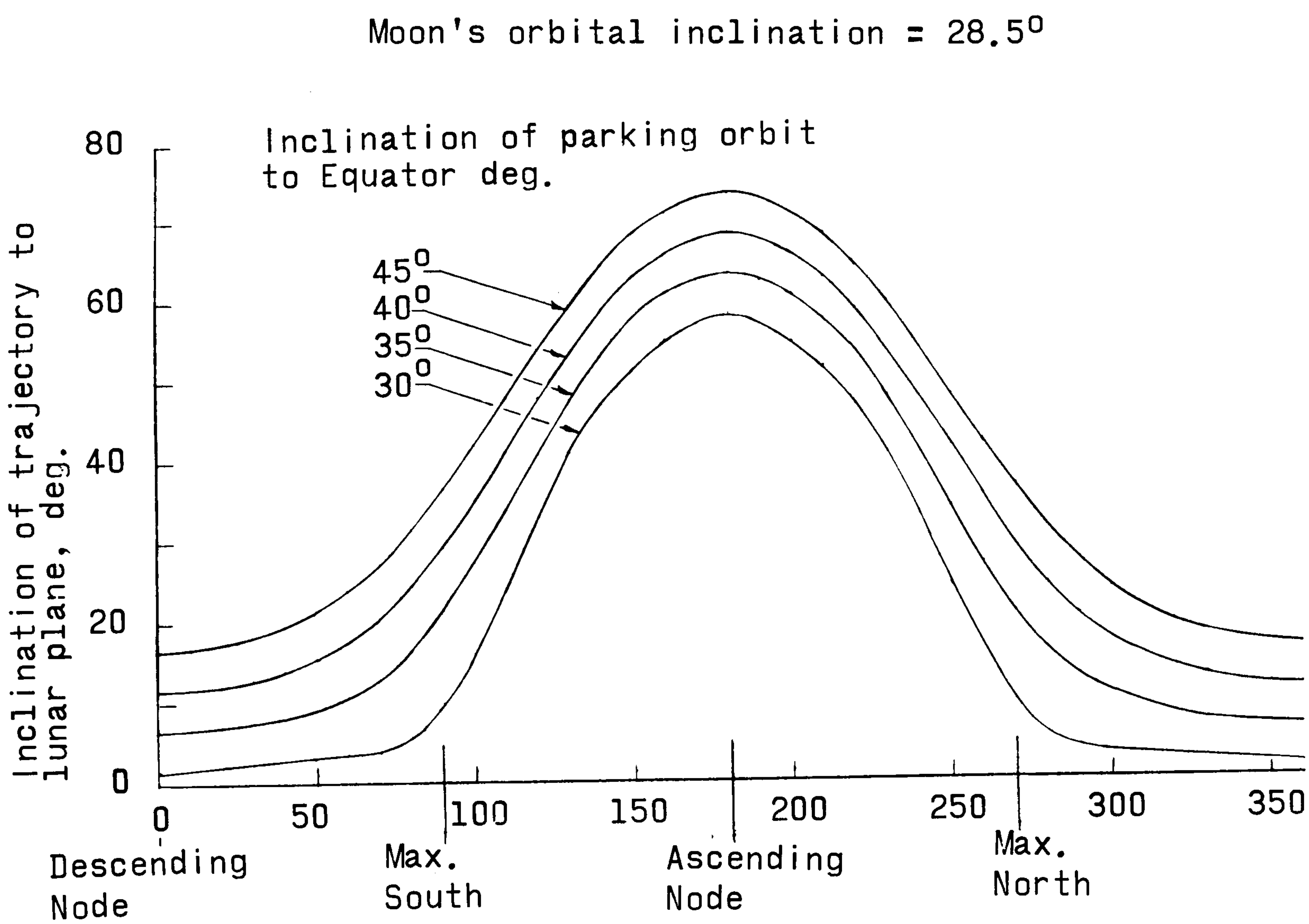


Figure 41.- Inclination of trajectory to moon's orbital plane and equatorial plane.

[redacted]

[redacted]

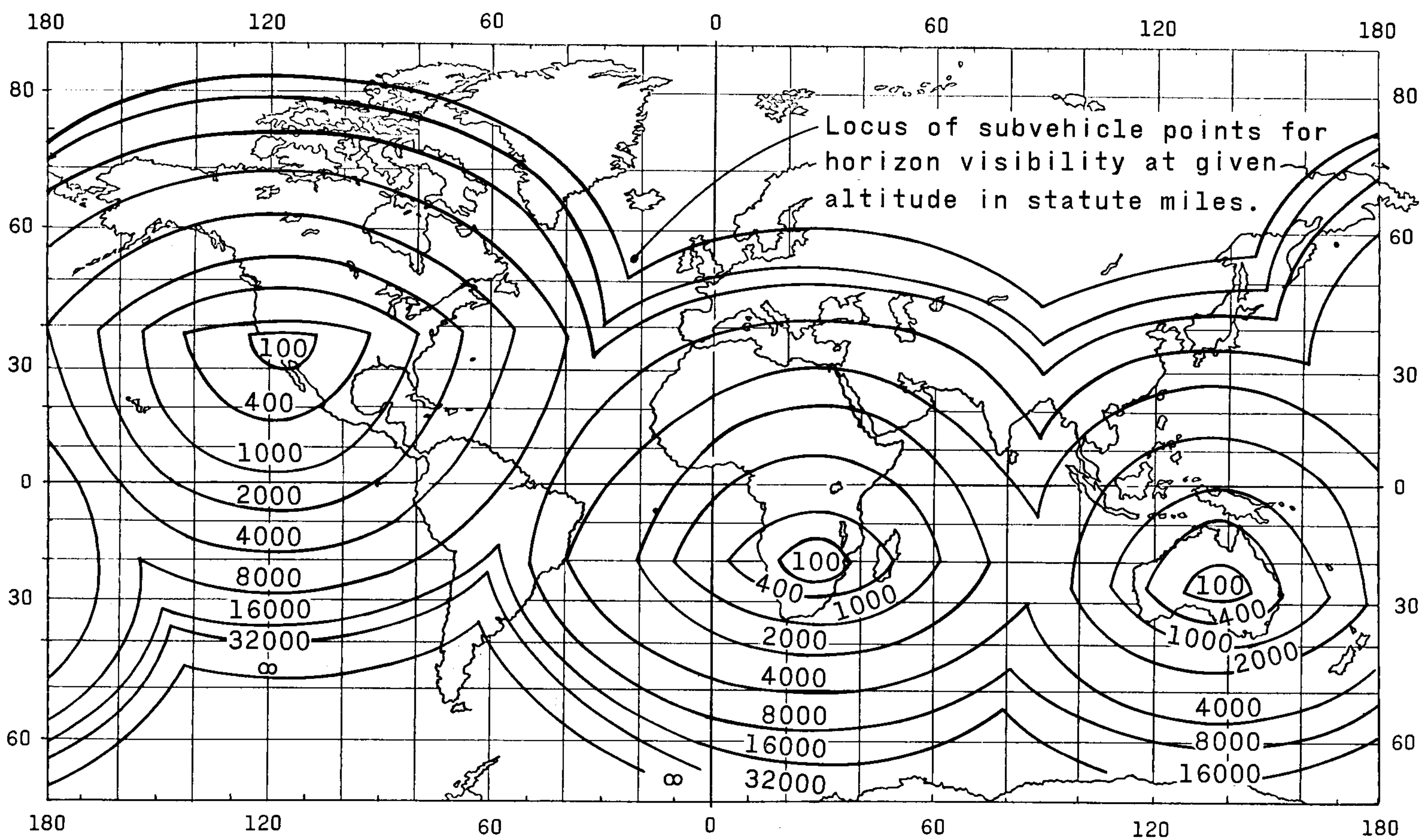


Figure 42.- Deep Space Station coverage plots for a 5° terrain horizon mask.

[redacted]

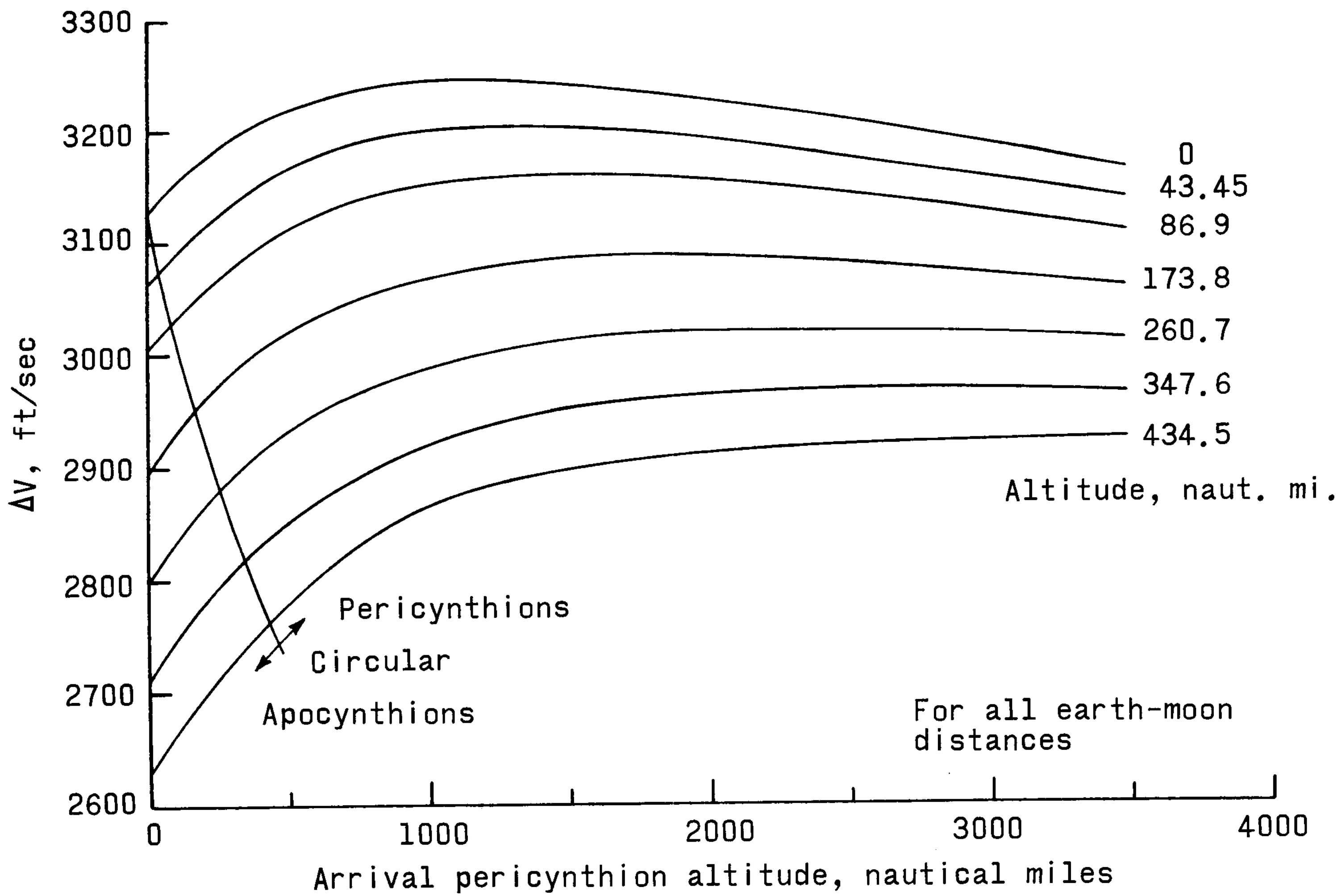


Figure 43.- Retro-velocity requirements to establish local lunar orbits from earth returning circumlunar trajectories.

[redacted]

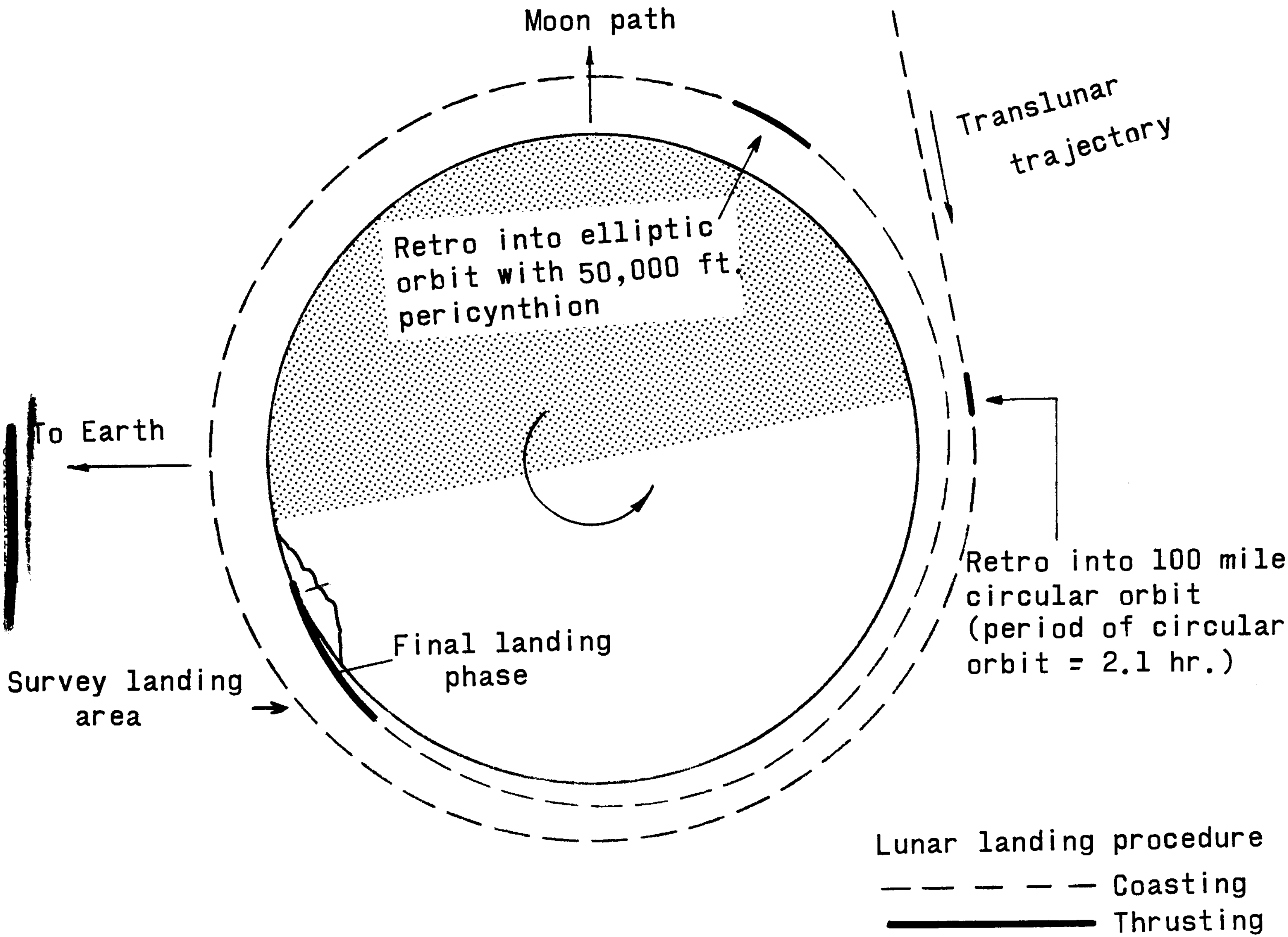
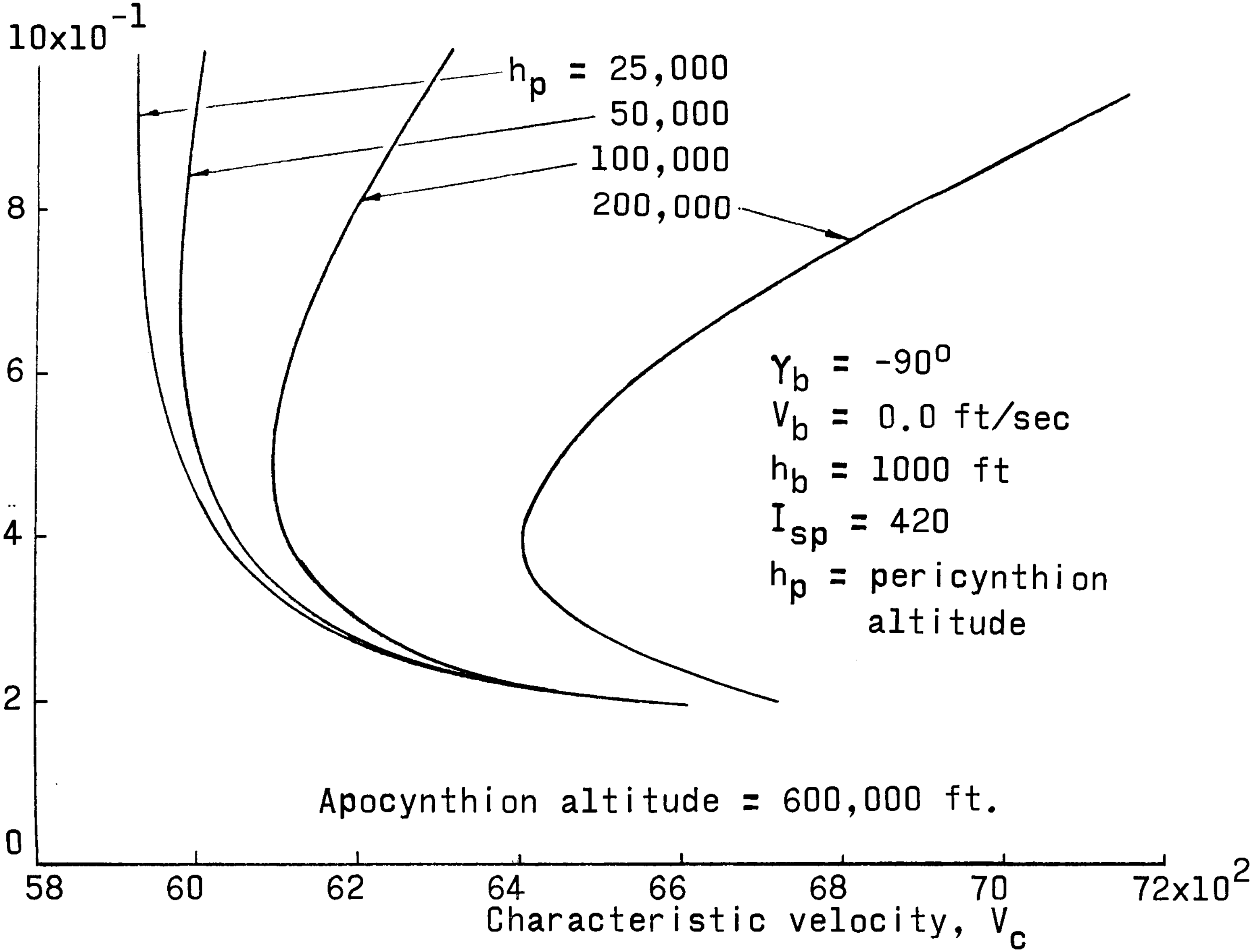


Figure 44.- Lunar landing technique.

[redacted]

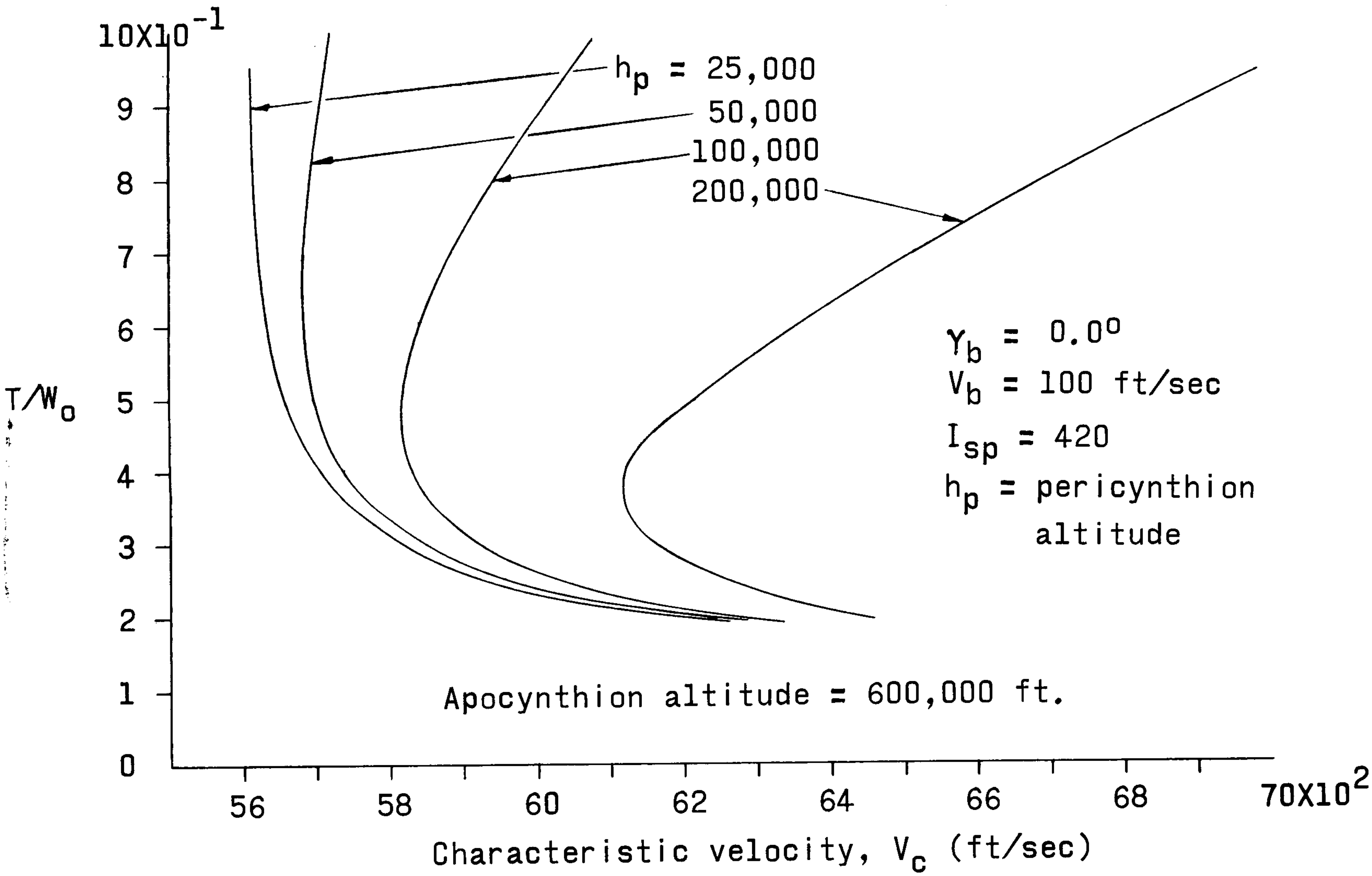
[redacted]



(a) Vertical lunar landing.

Figure 45.- Characteristic velocity for optimum lunar landings from elliptic orbits with various pericynthion altitudes.

[redacted]



(b) Horizontal lunar landing.

Figure 45.- Concluded.

[redacted]

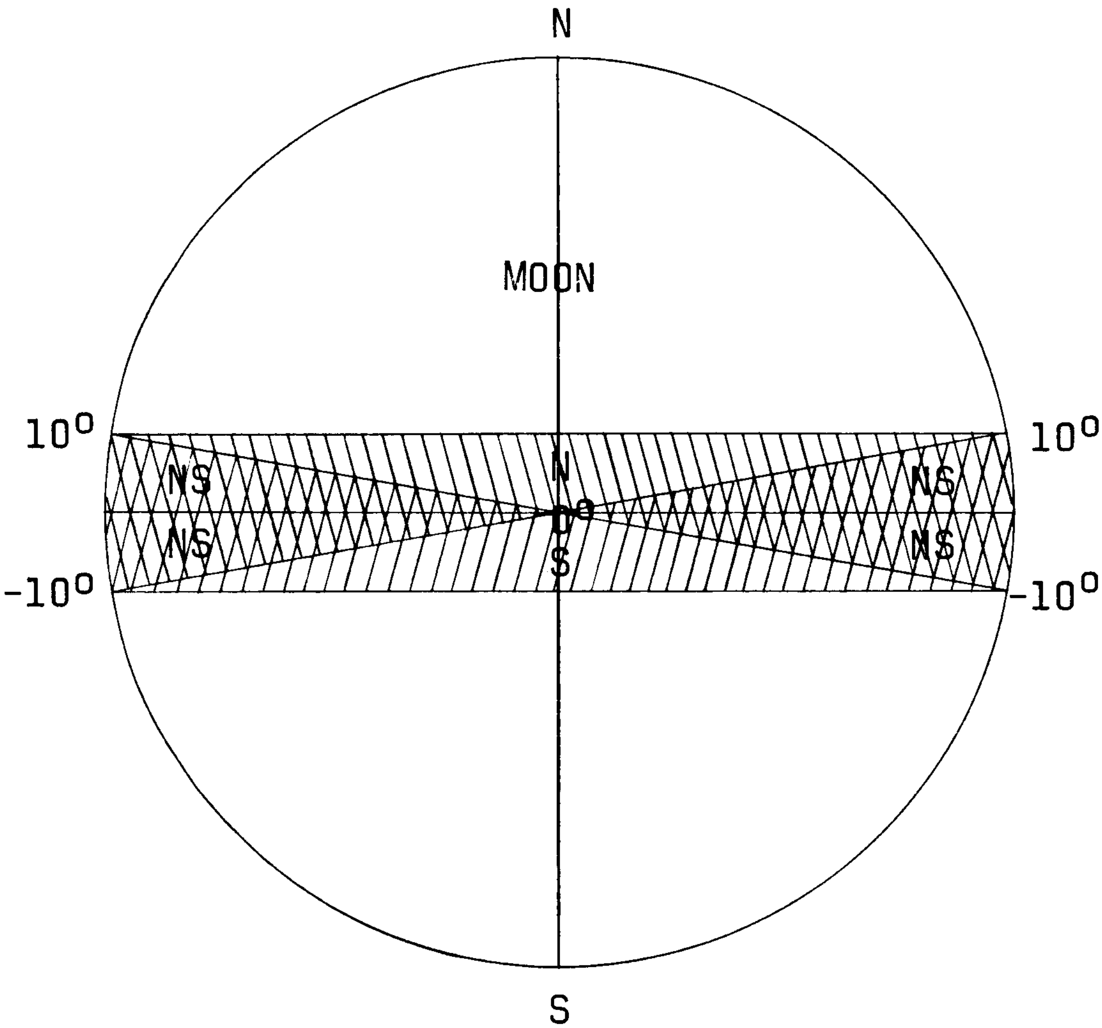


Figure 46.- Landing area for lunar mission
with free return translunar
trajectories - no plane changes.

[redacted]

[redacted]

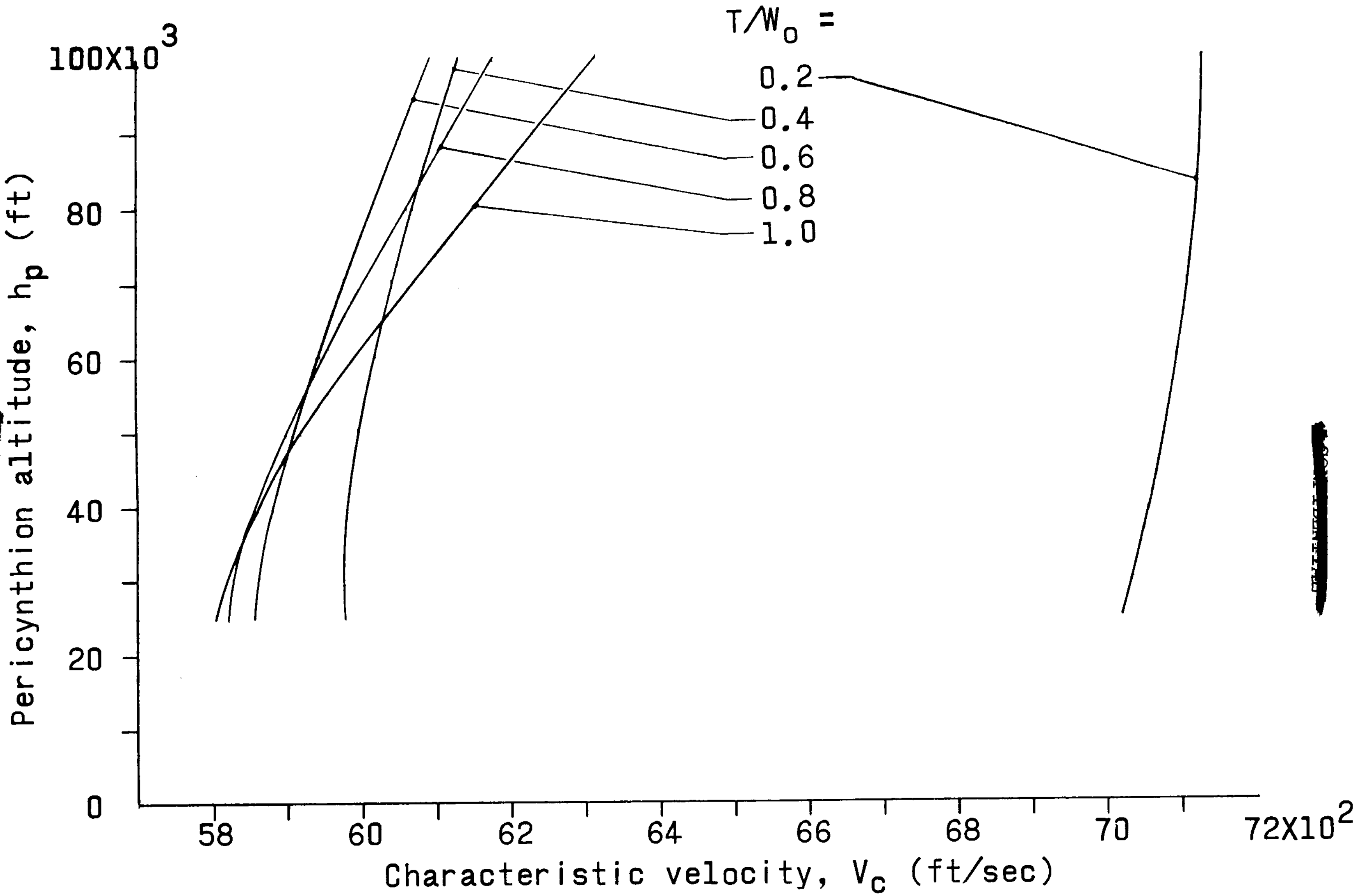


Figure 47.- Characteristic velocity for lunar launches to elliptic orbits with various T/W_o. Apocynthion altitude = 600,000 ft.

[redacted]

[redacted]

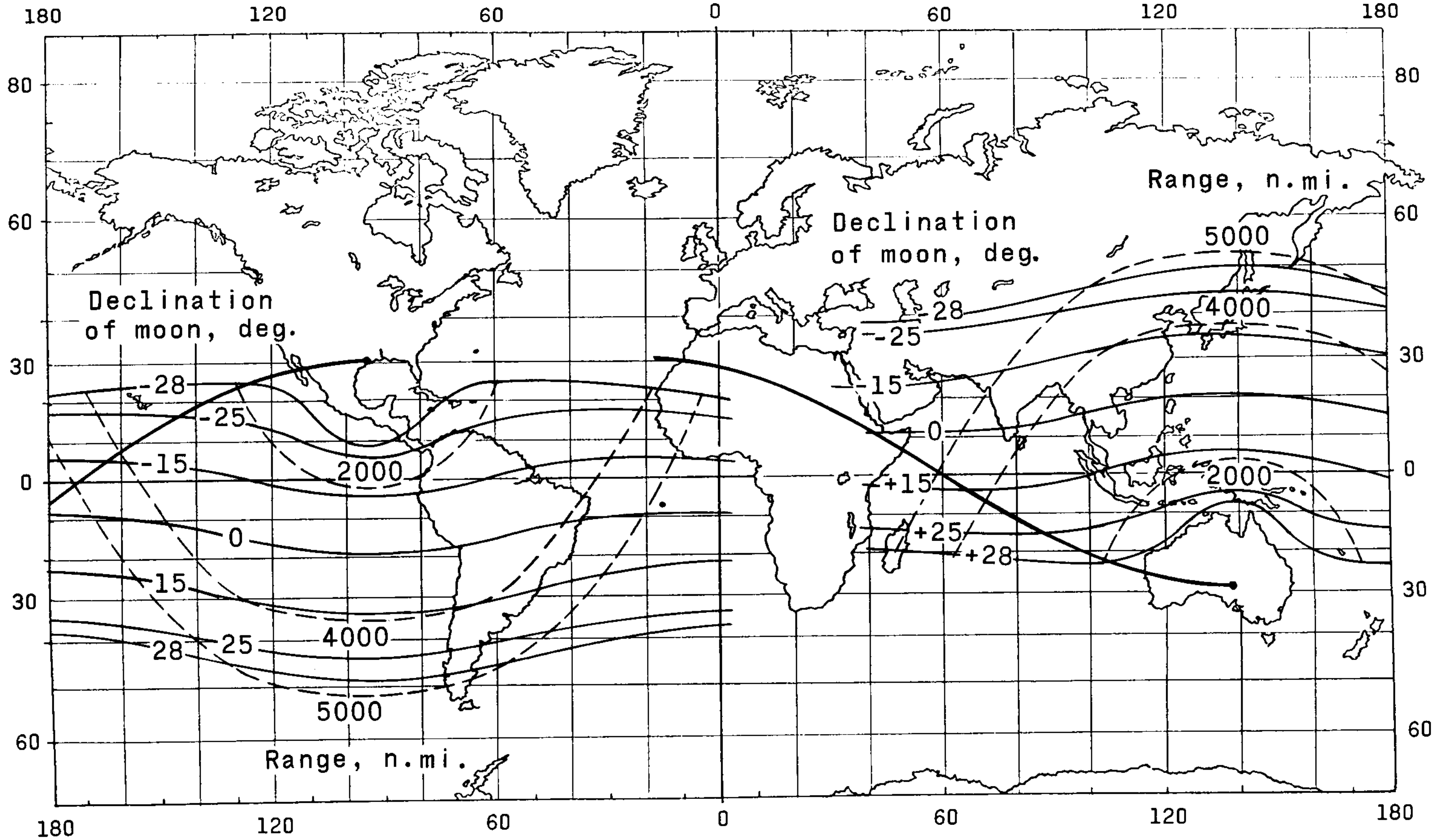
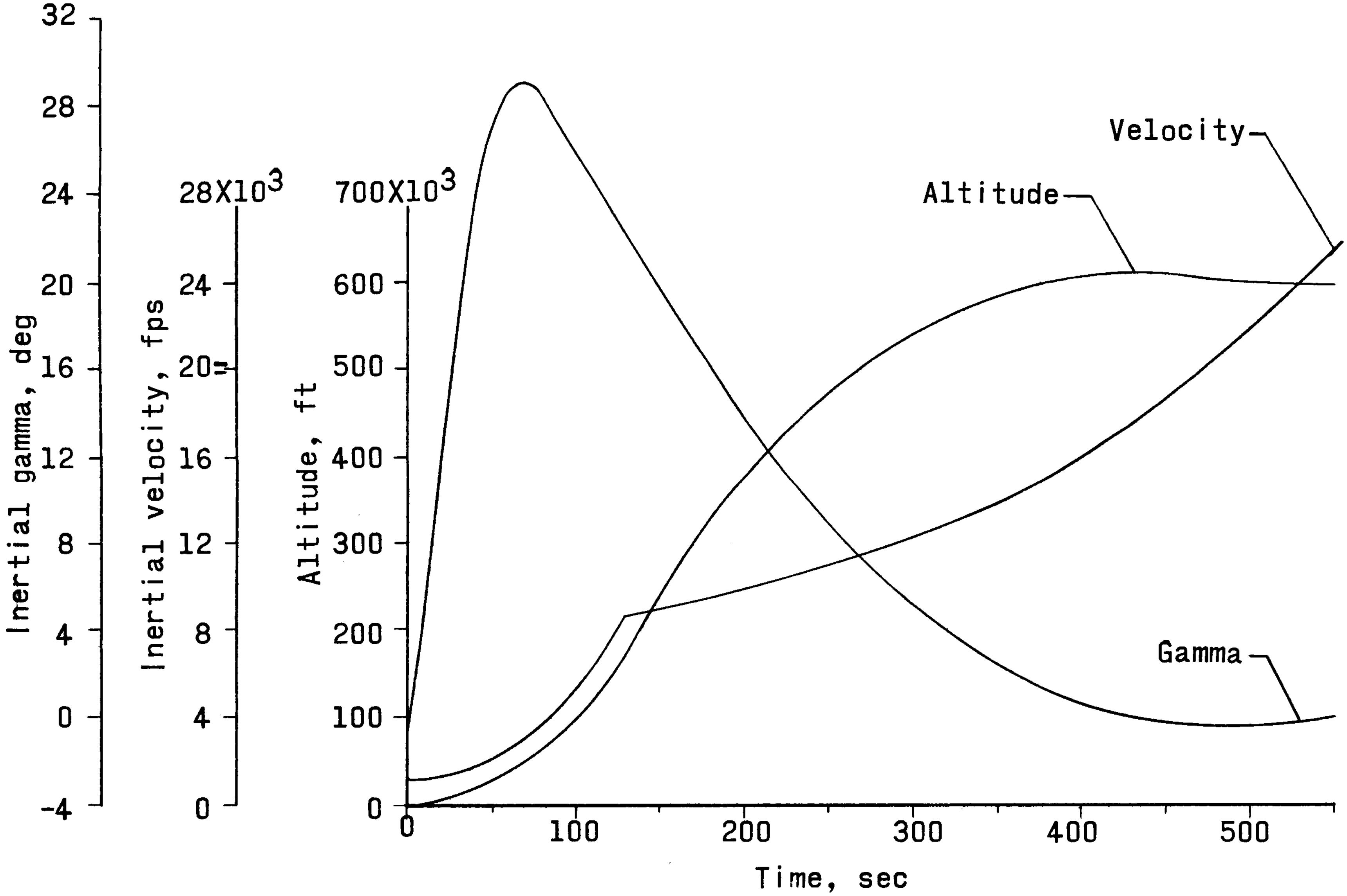


Figure 48.- Locus of reentry points for landing sites in U.S.A. and Australia.

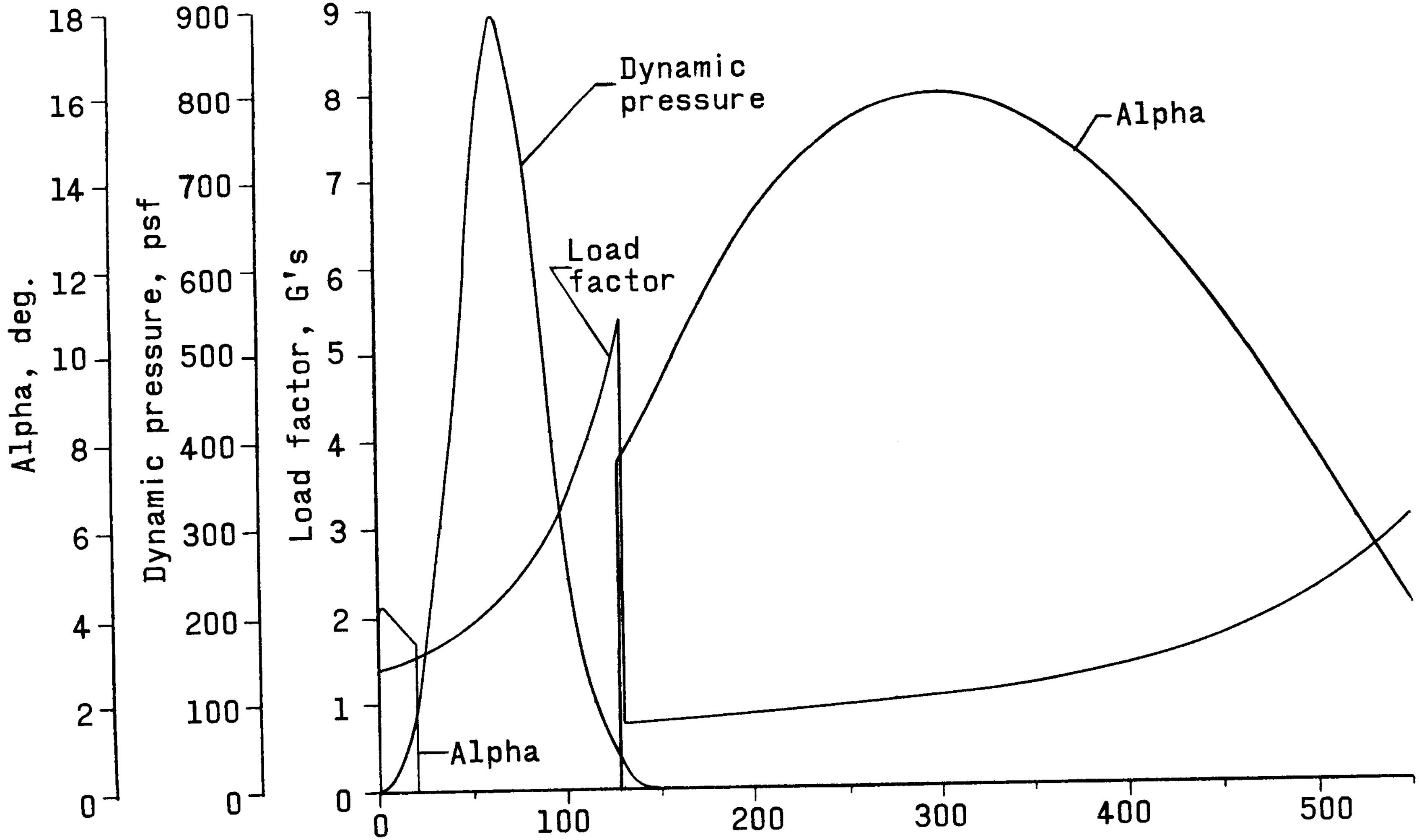
[redacted]



(a) Flight path.

Figure 49.- Time history from lift-off to parking orbit.

[redacted]



(b) Loads.

Figure 49.- Concluded.

[redacted]

[redacted]

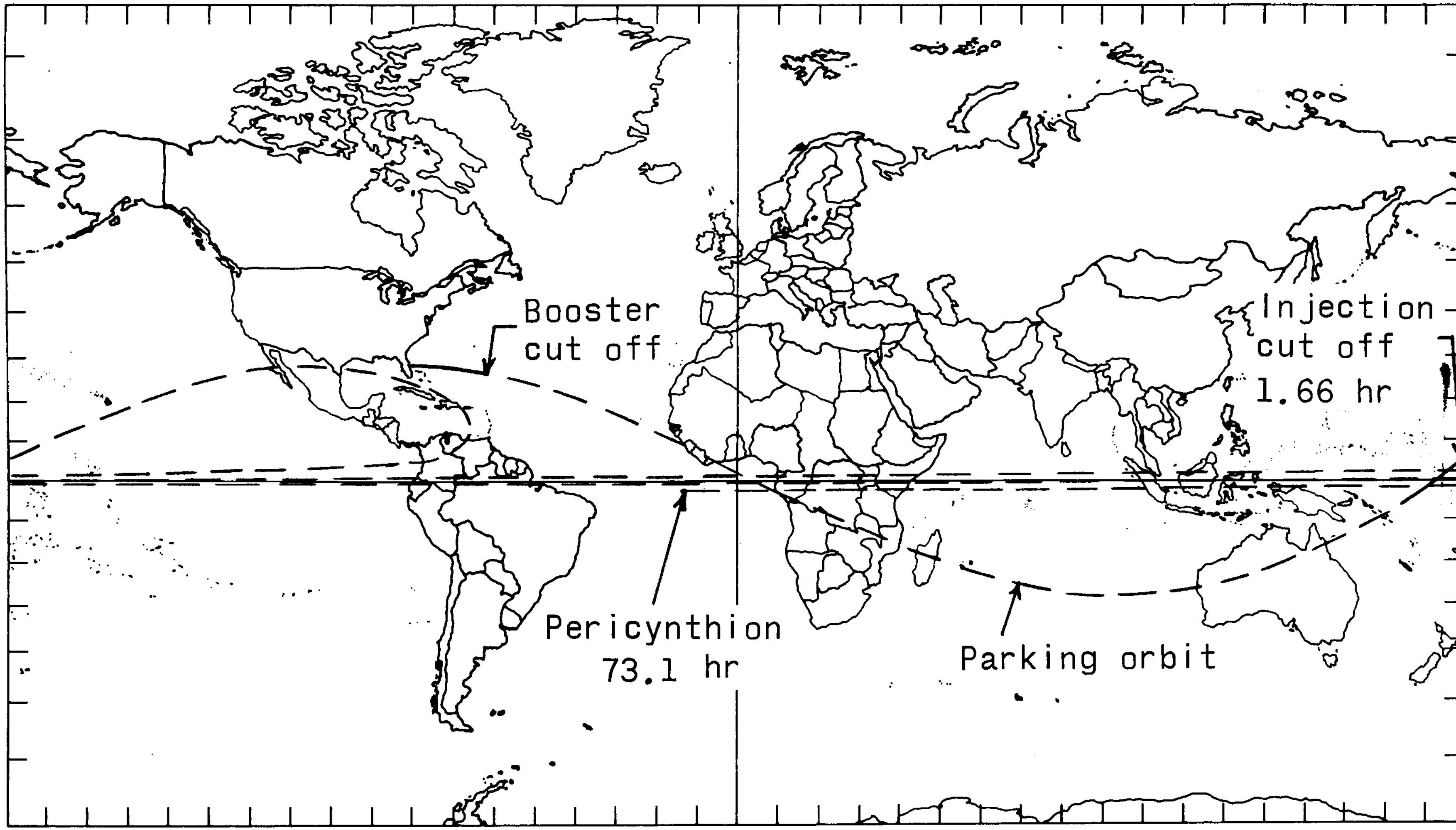


Figure 50.- Earth track for lunar flight plan.

[redacted]

[redacted]

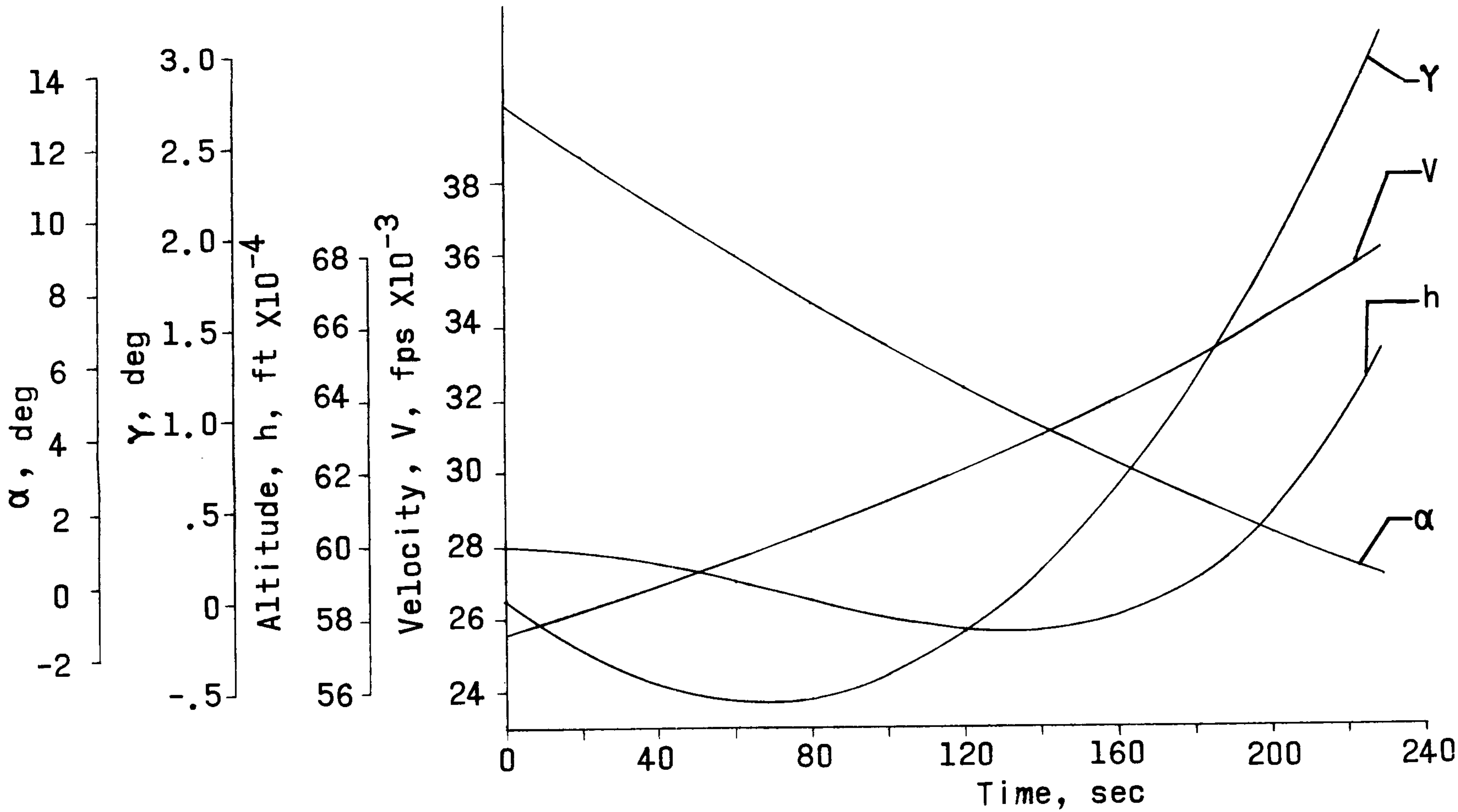


Figure 51.- Time history of transfer from parking orbit to translunar trajectory.

[redacted]

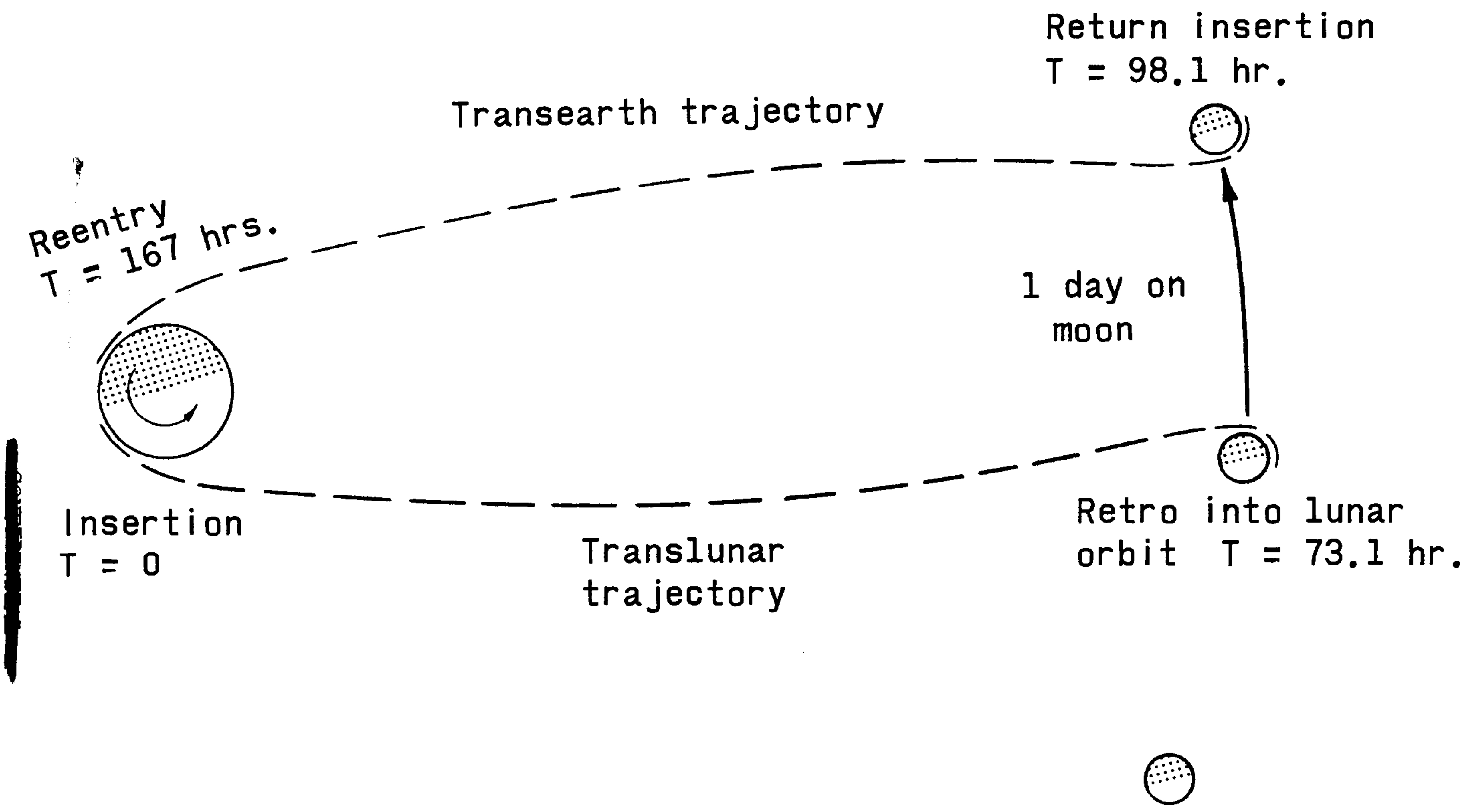


Figure 52.- Translunar and transearth trajectories shown in the inertial Earth-Moon system.

[redacted]

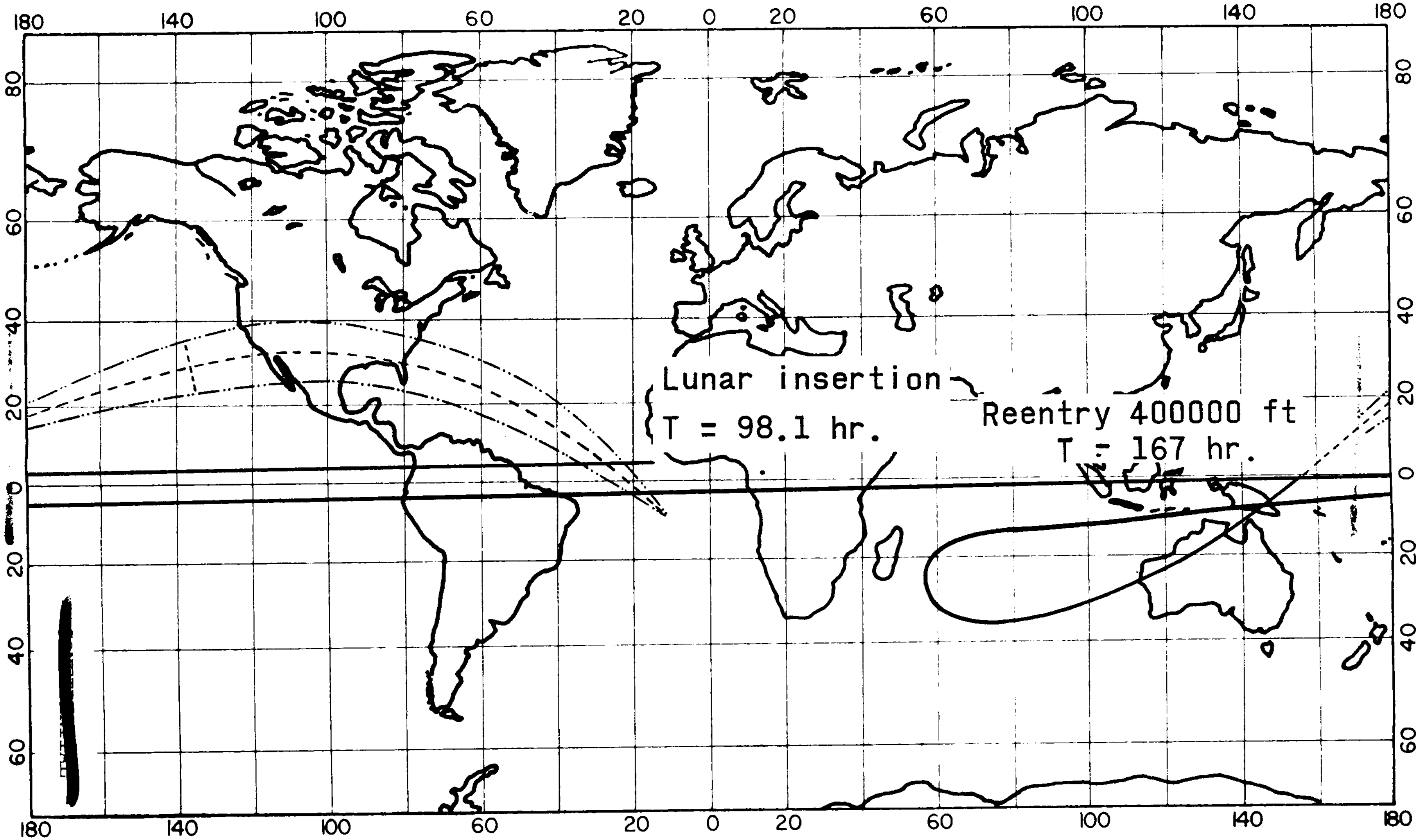


Figure 53.- Earth track of transearth and reentry phases.

[redacted]

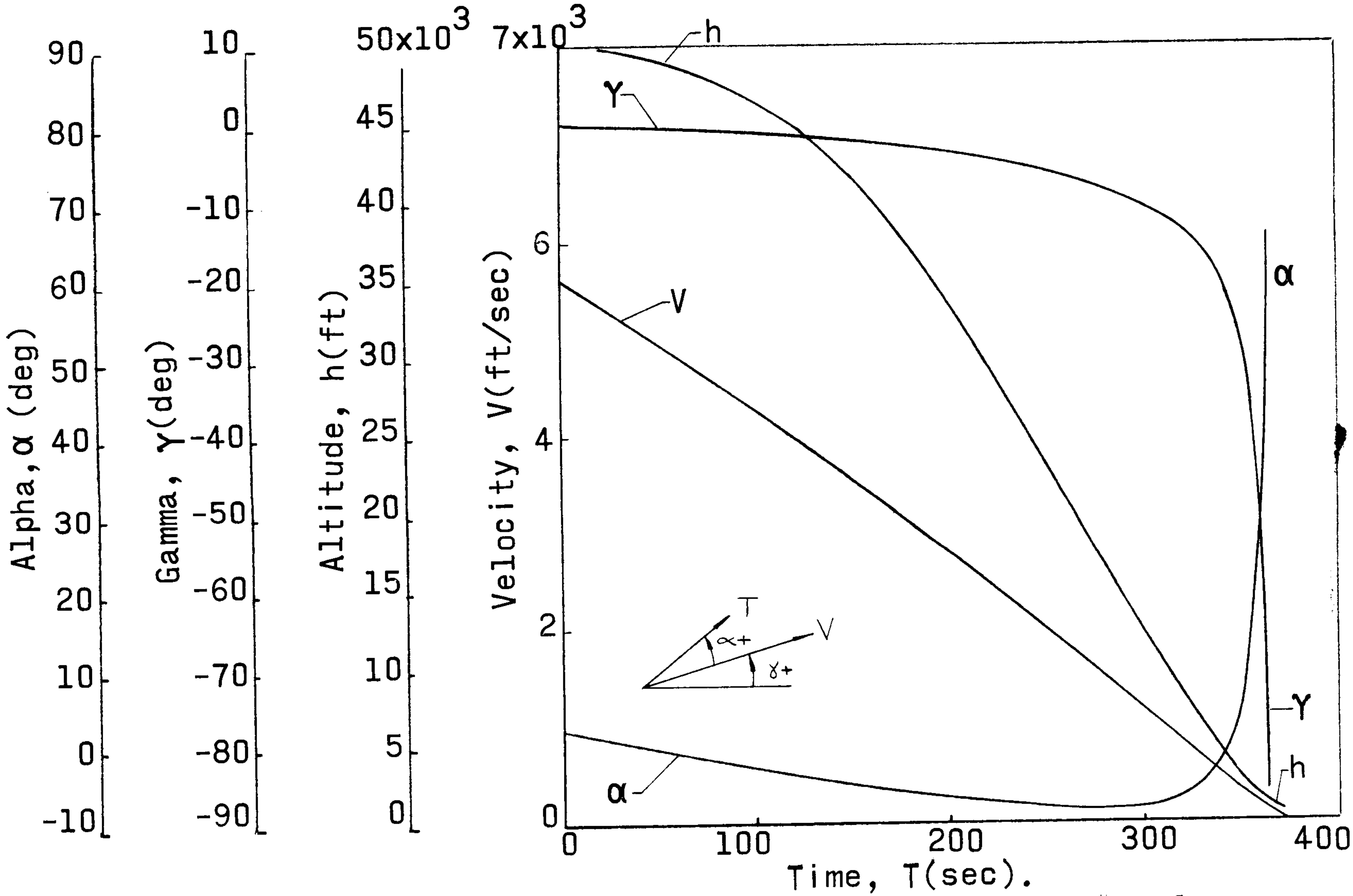


Figure 54.- Time history for optimum vertical lunar landing from 50,000 ft pericynthion. $\gamma_b = -90^o$, $V_b = 0.0$ fps, $h_b = 650$ ft, $T/W_o = 0.4$, Isp = 420. Apocynthion altitude = 600,000 ft.

[redacted]

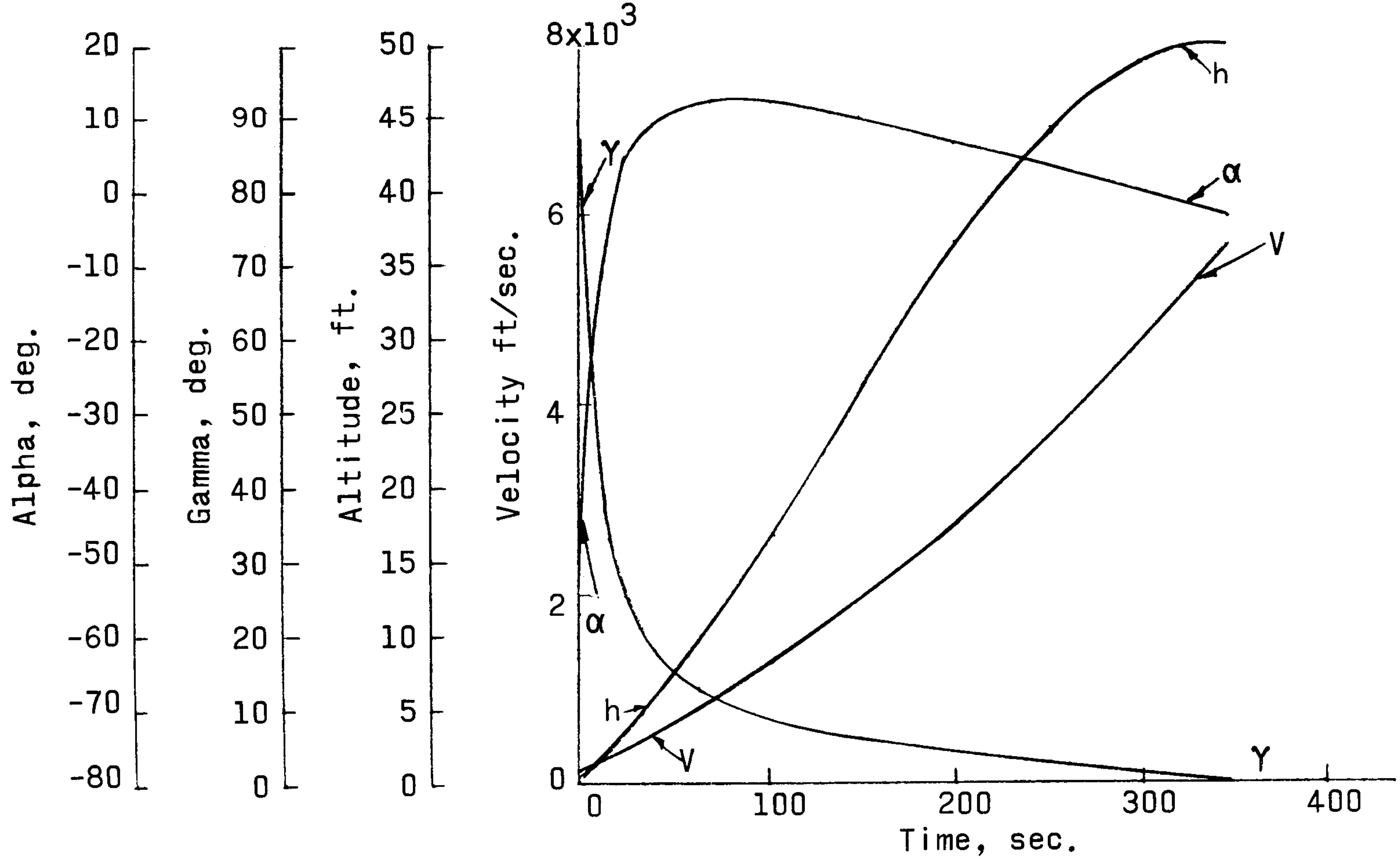


Figure 55.- Time history for optimum lunar launch to 50,000 ft. pericynthion. $\gamma_b = 0.0^o$, $V_b = V_p$, $T/W_o = 0.4$, $I_{sp} = 305$, apocynthion altitude = 600,000 ft

[redacted]

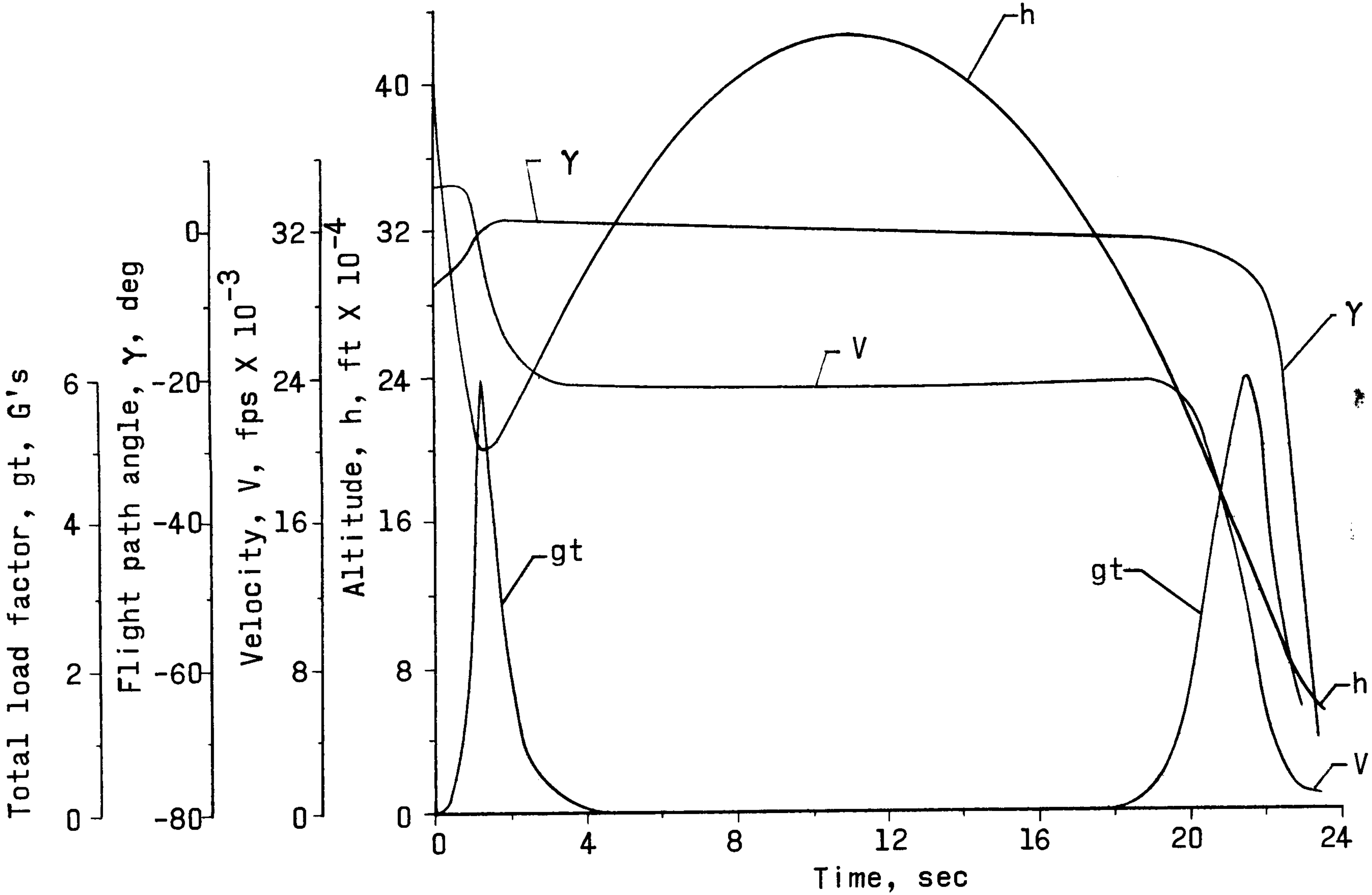


Figure 56.- Time history from reentry to near-landing.

[redacted]

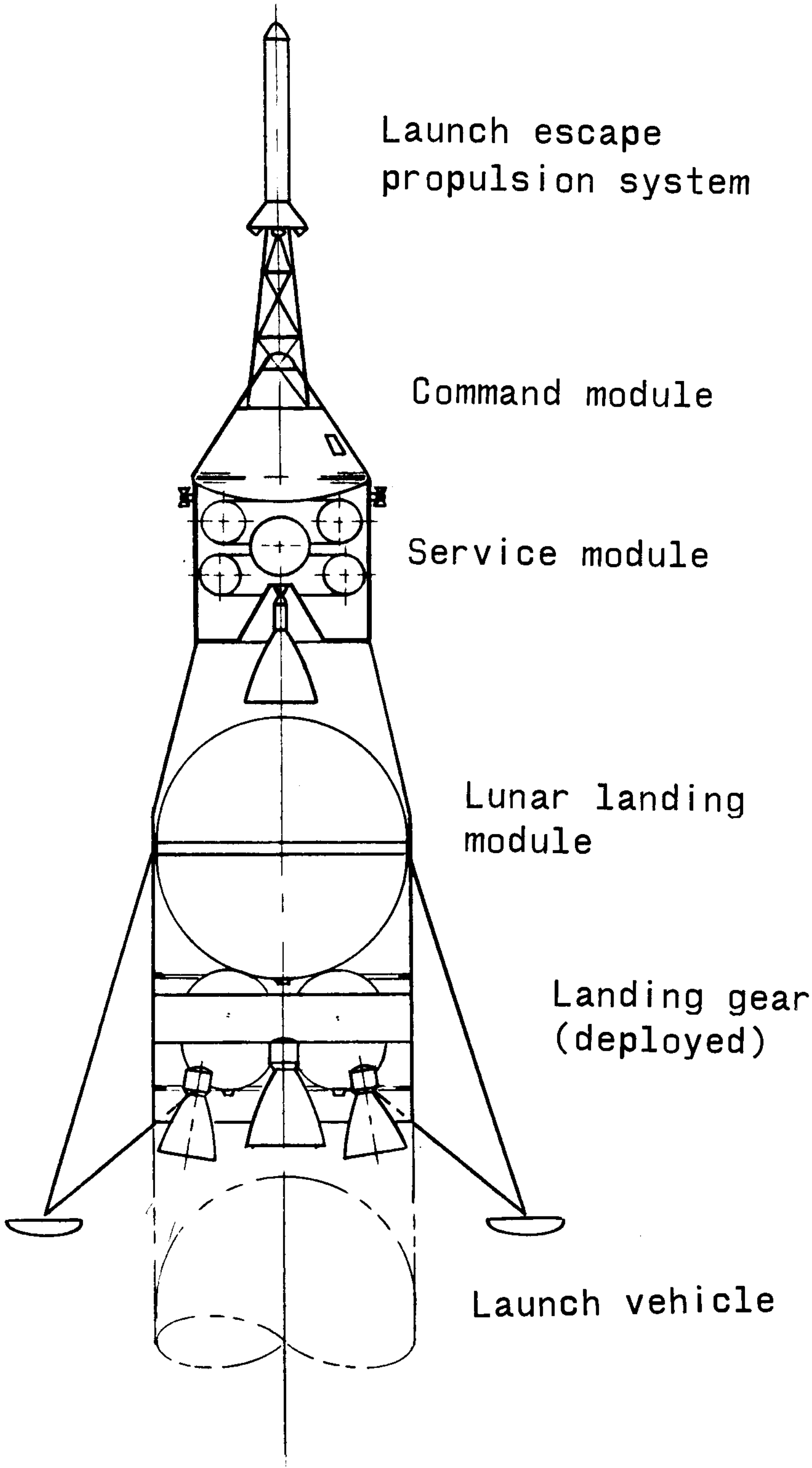


Figure 57.- General arrangement -
lunar landing configuration.

[redacted]

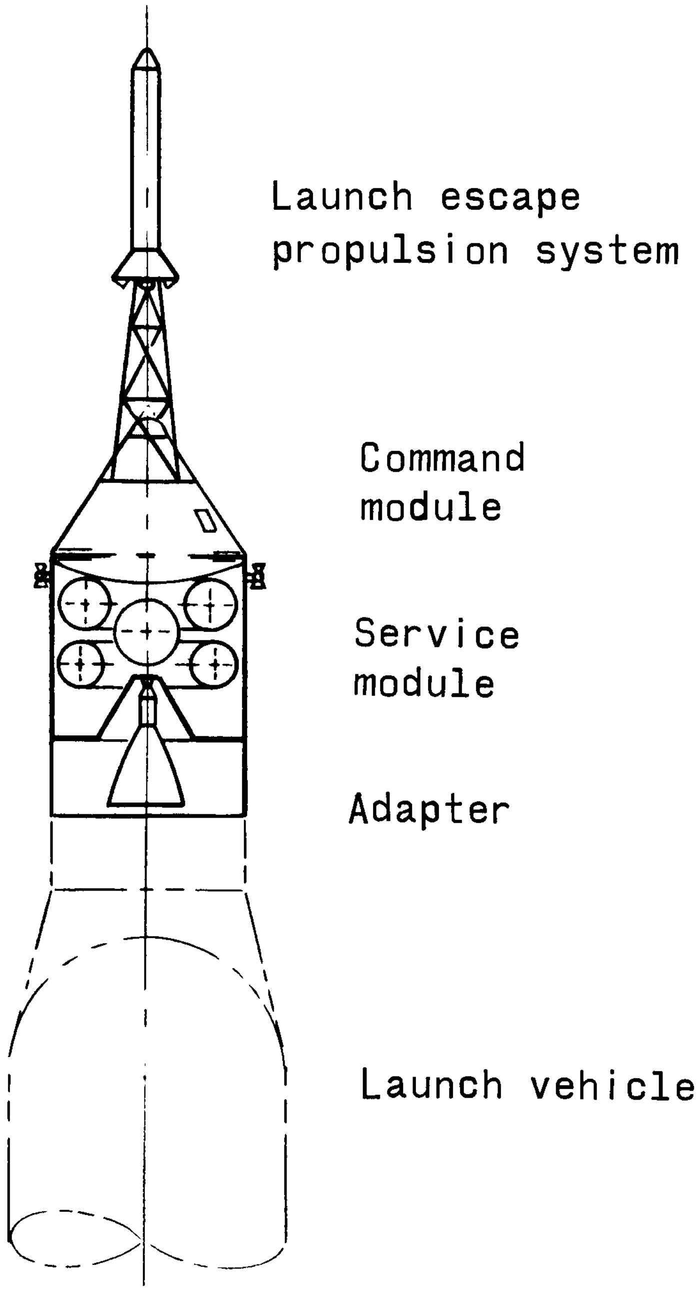


Figure 58.- General arrangement - earth orbital configuration.

[redacted]

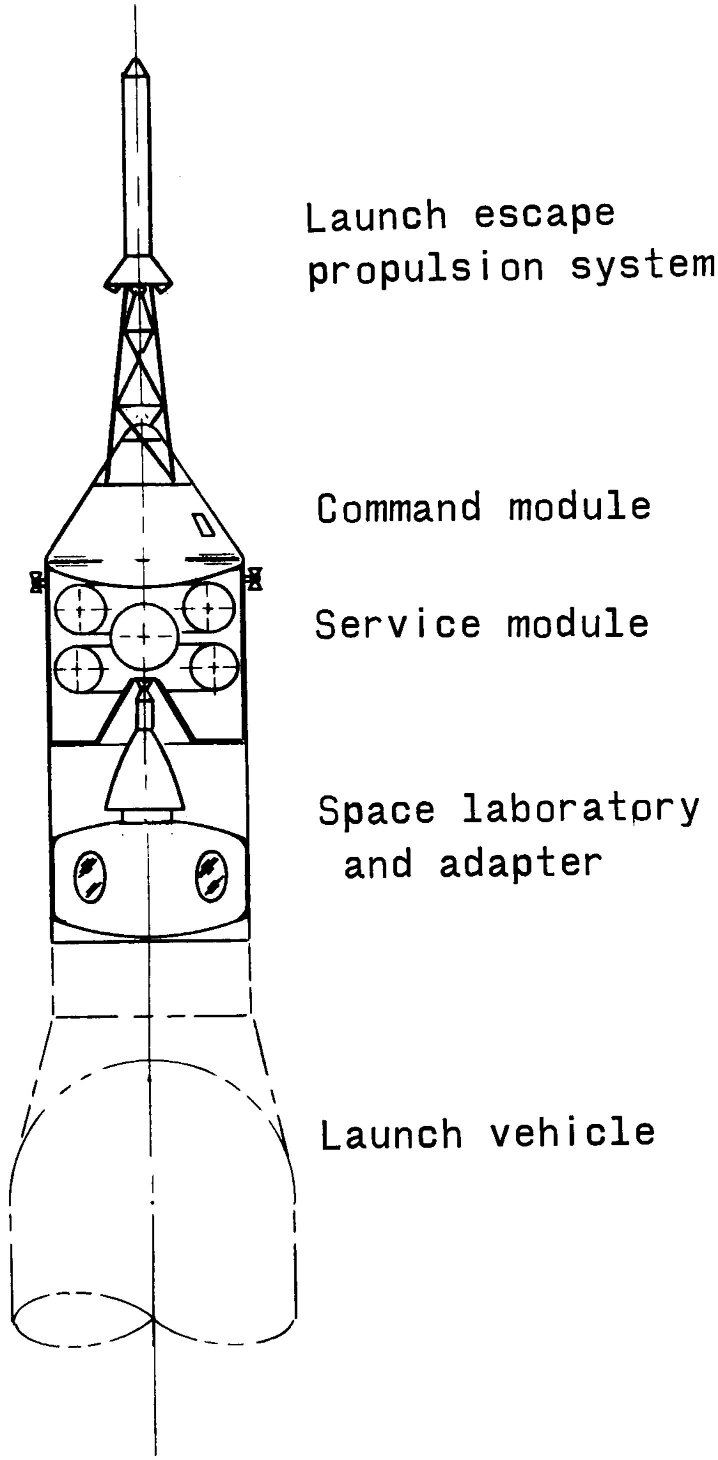


Figure 59.- General arrangement - earth orbital configuration with space laboratory.

[redacted]

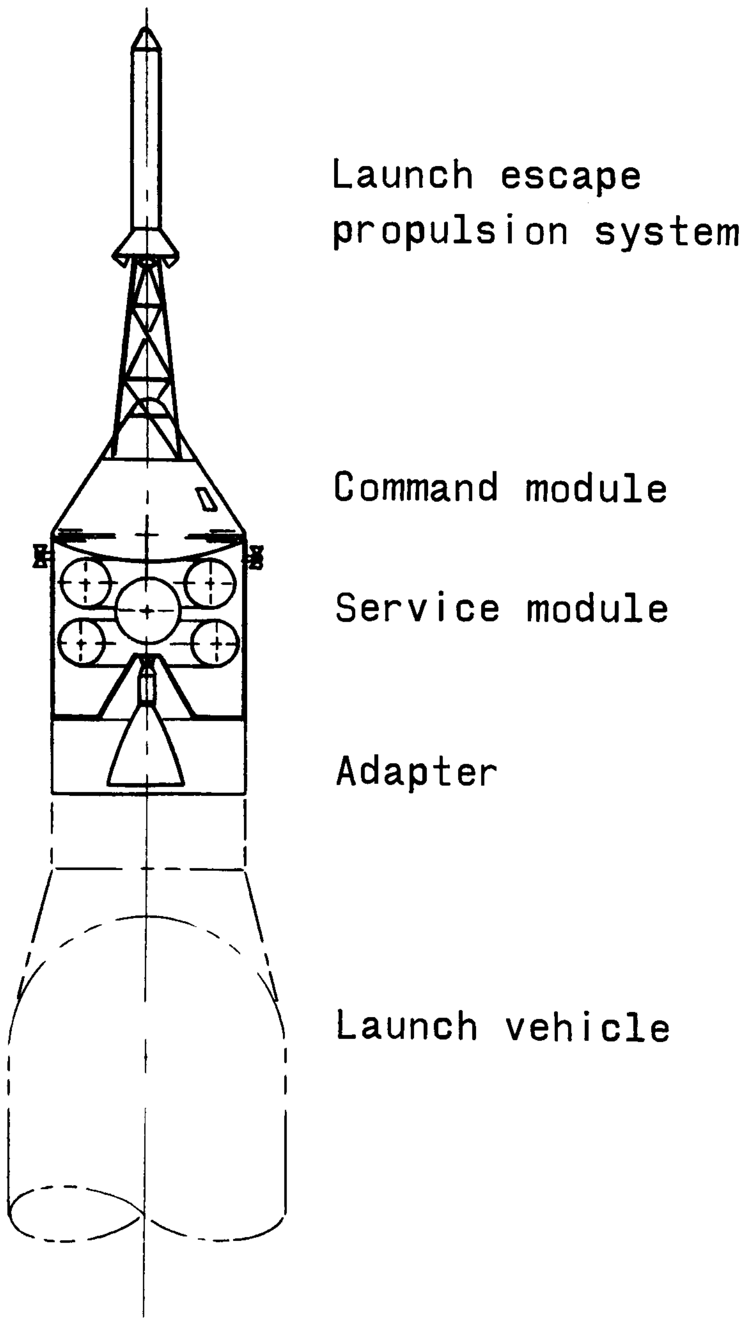


Figure 60.- General arrangement - circumlunar configuration.

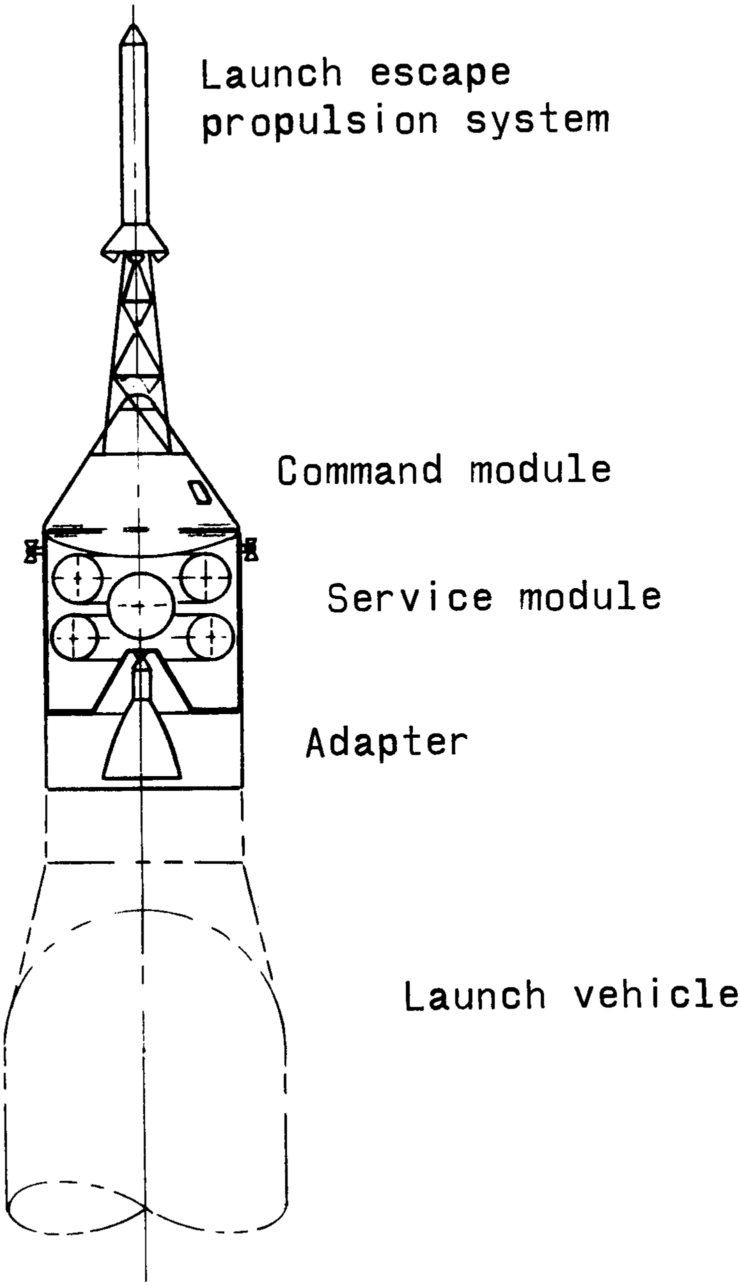


Figure 61.- General arrangement -
lunar orbital configuration.

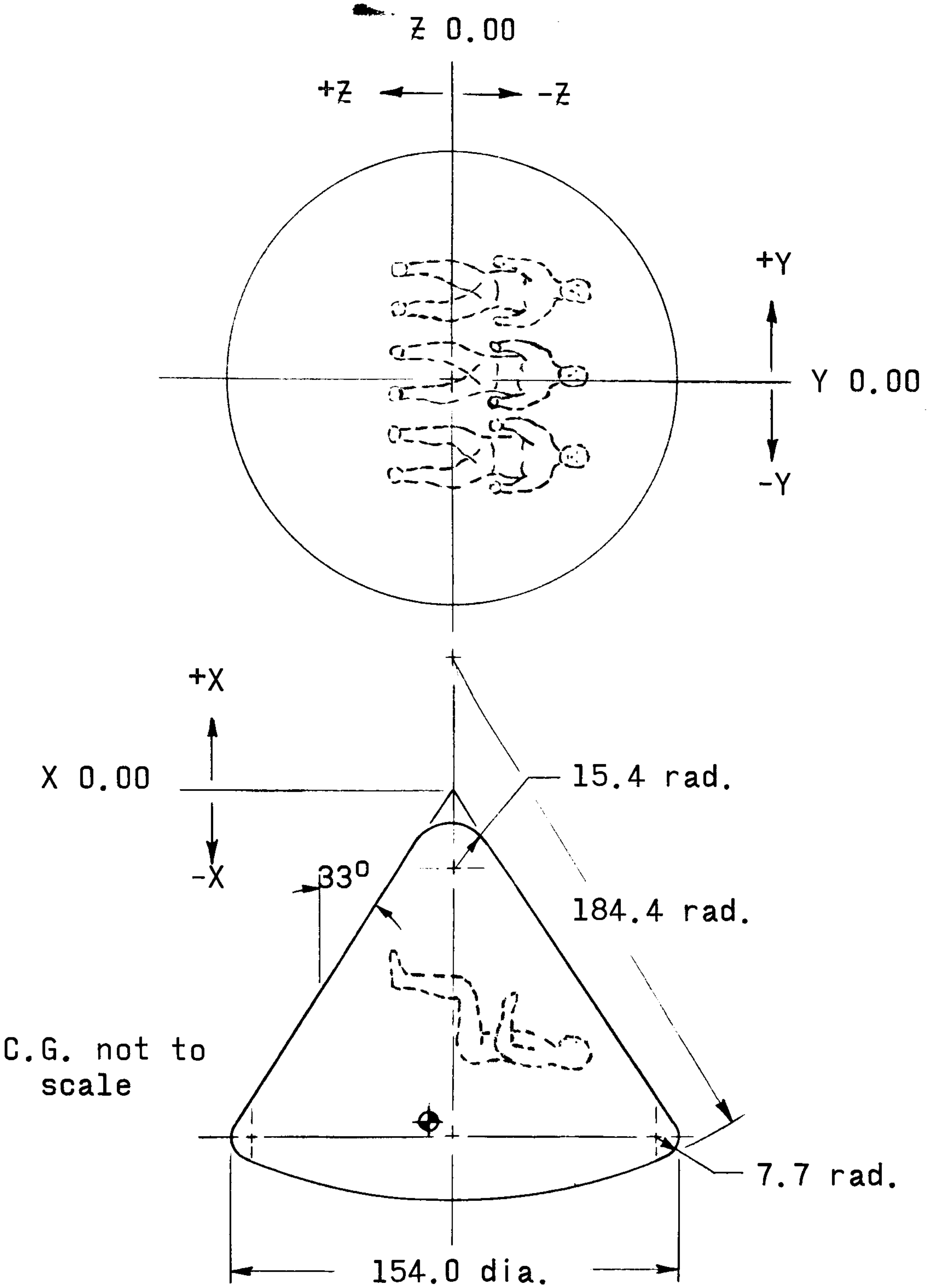


Figure 62.- Command module nominal geometry.

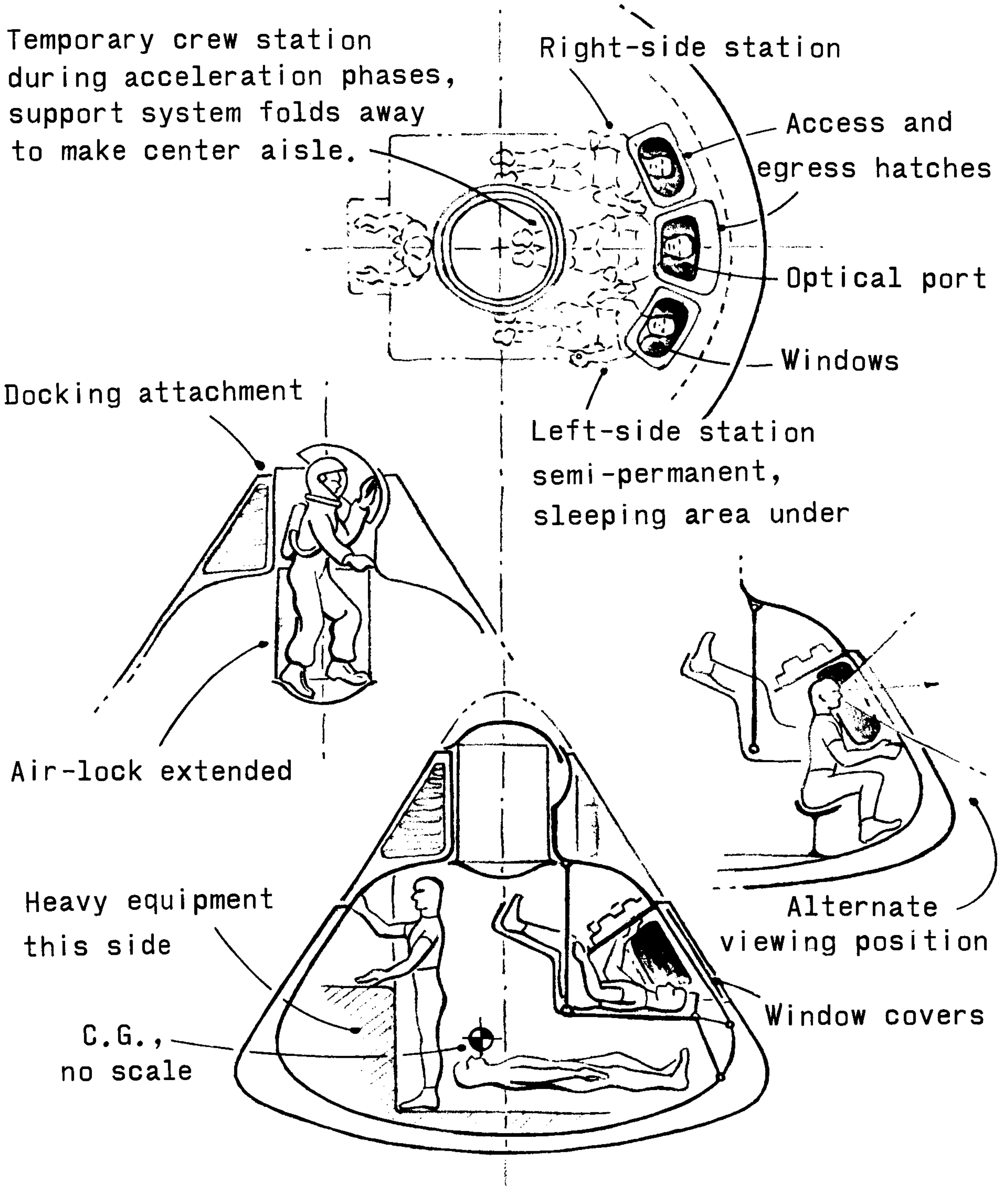


Figure 63.- Command module - Inboard profile, activity areas.

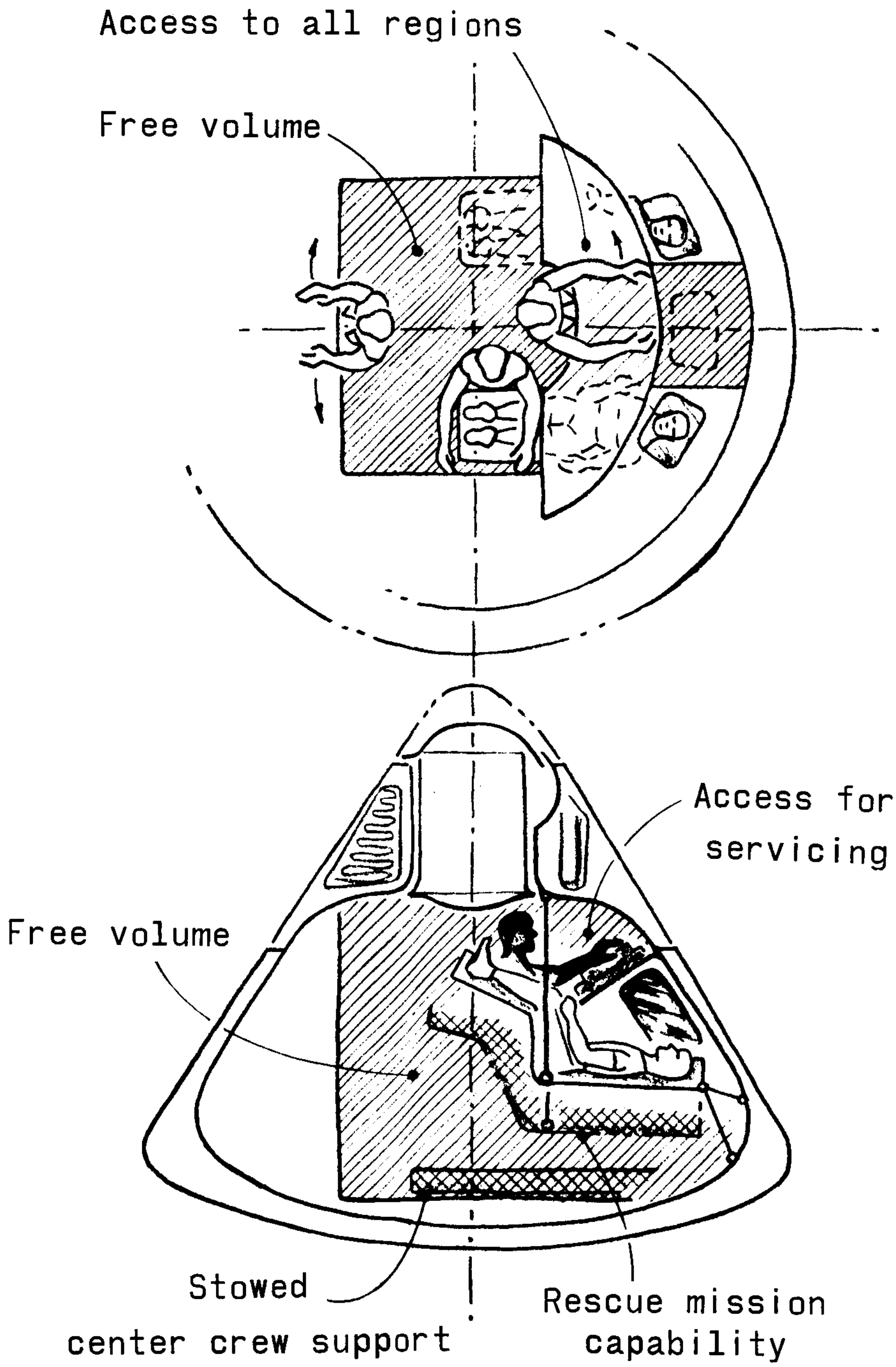


Figure 64.- Command module - Inboard profile, volume utilization.

[redacted]

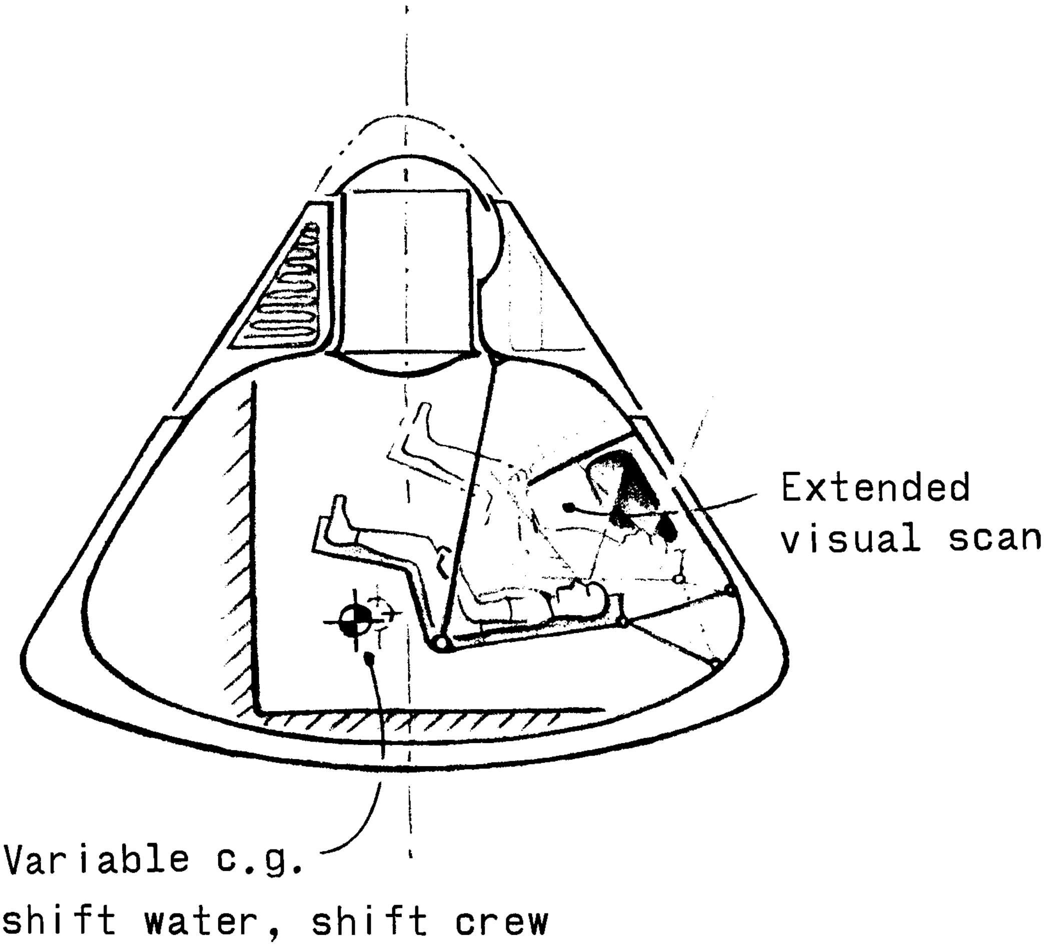


Figure 65.- Command module - Inboard profile,
display coverage and center of
gravity control.

[redacted]

2T

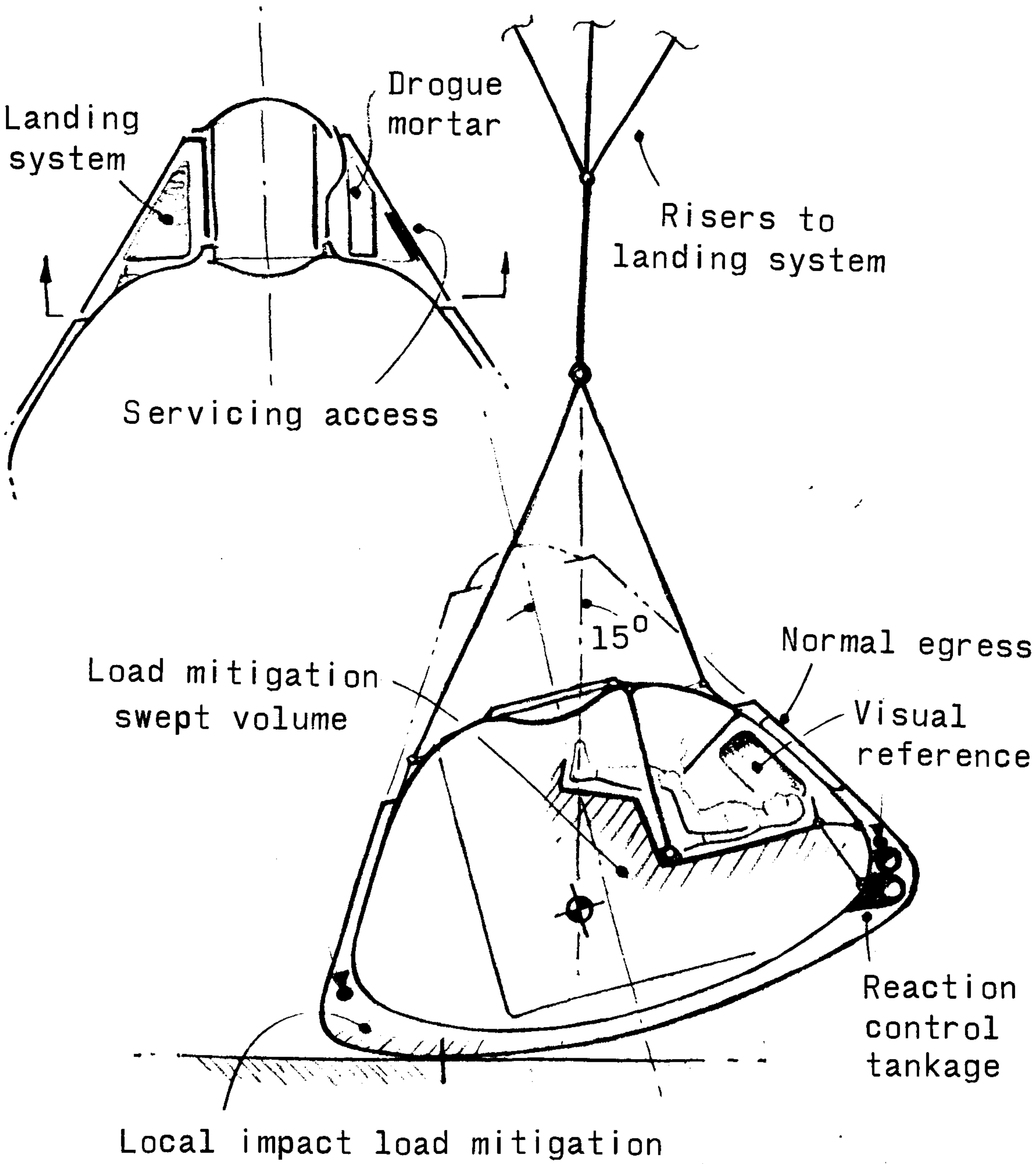


Figure 66.- Command module - Inboard profile, landing considerations.

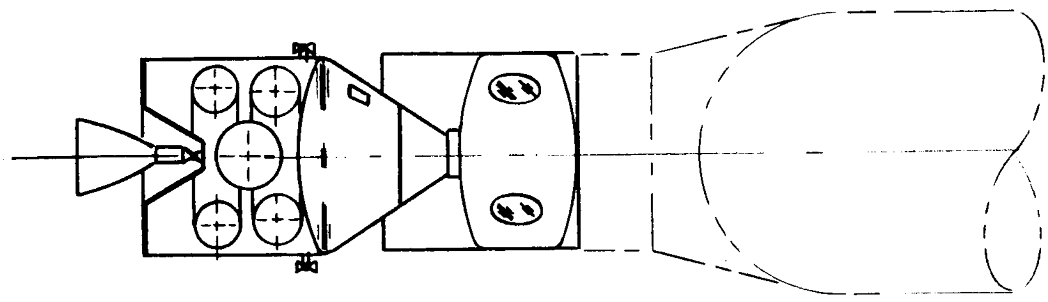


Figure 67.- Docking arrangement - space laboratory.

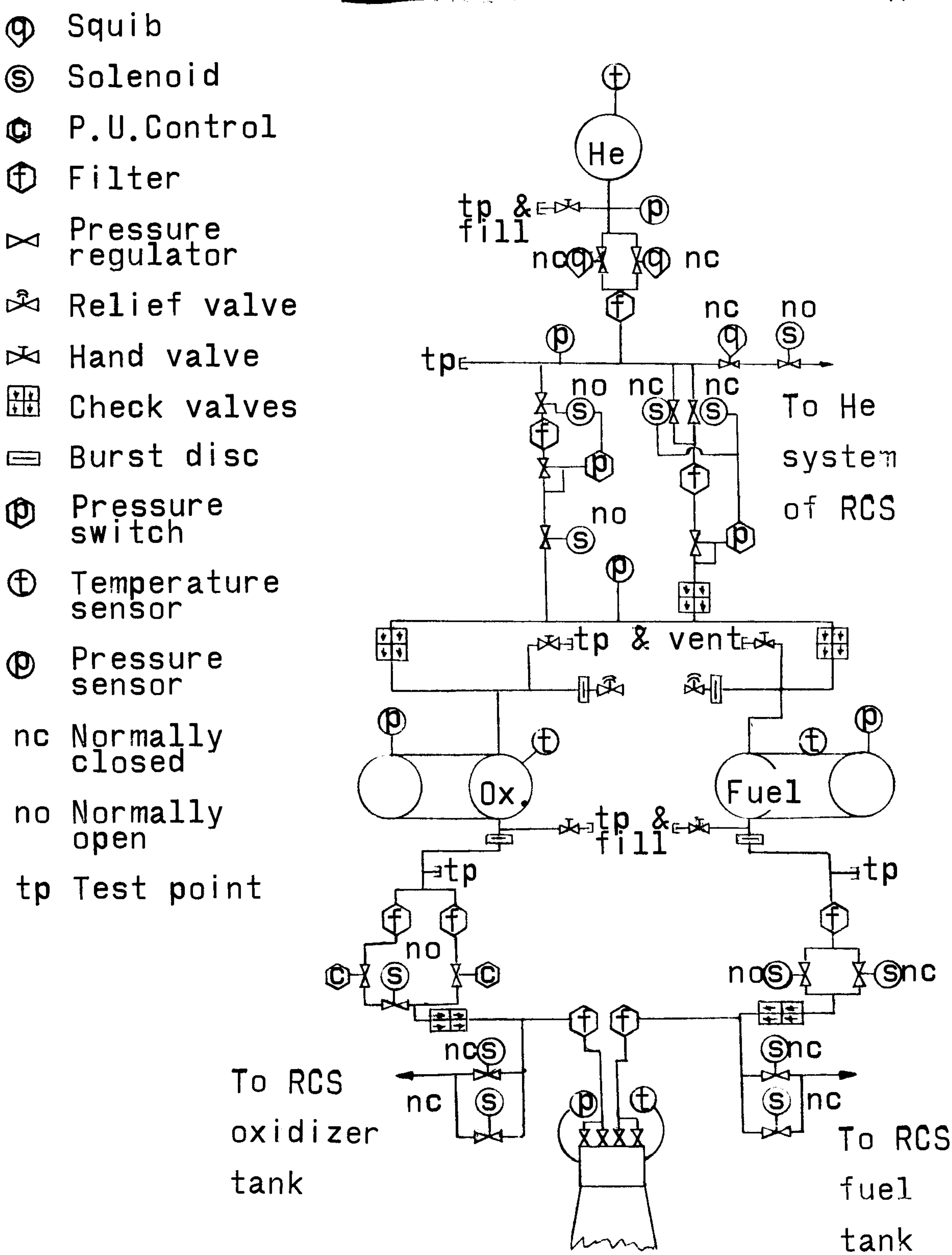


Figure 68.-Service propulsion system, Single chamber

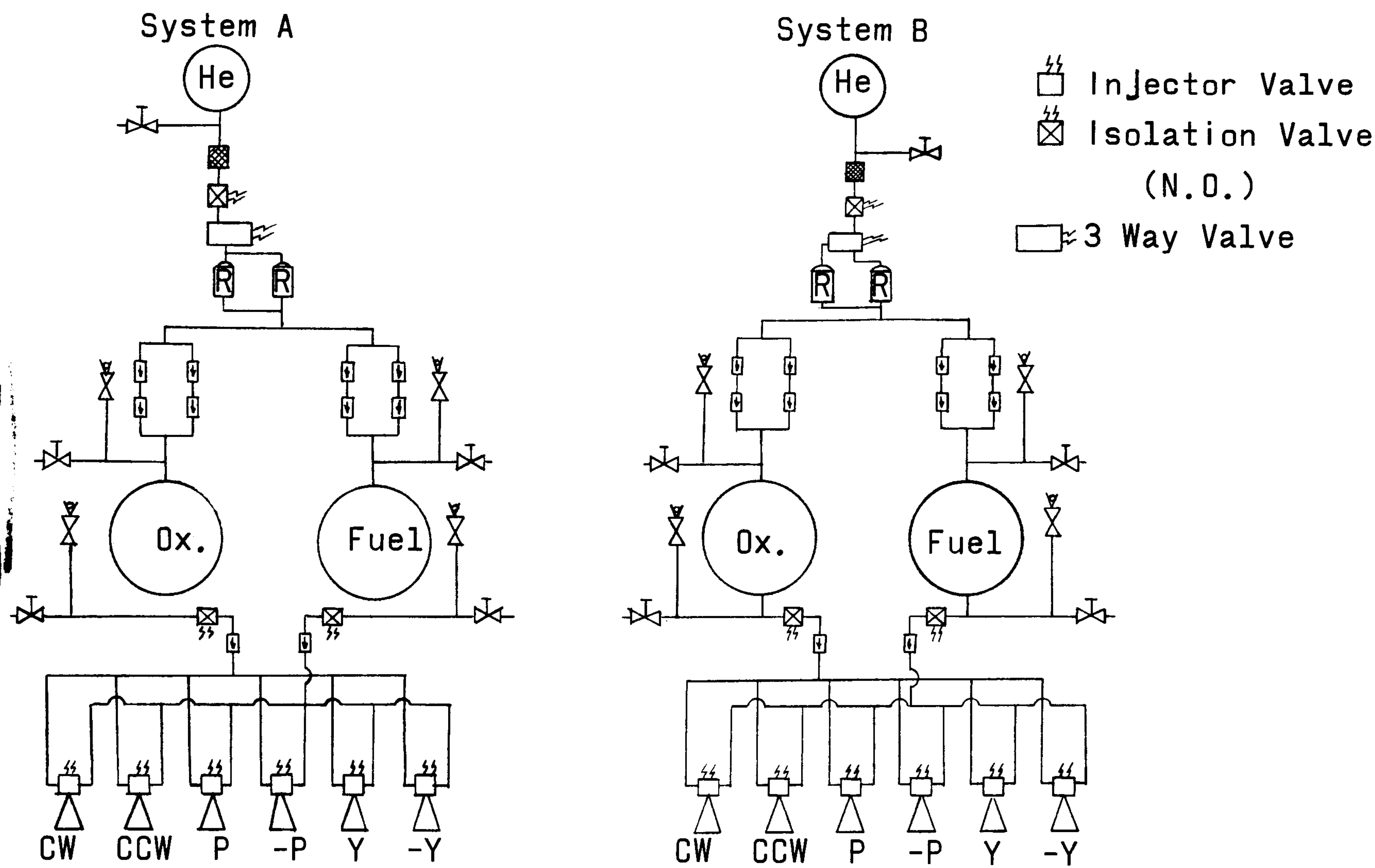


Figure 69.- Reaction control system, command module, schematic.

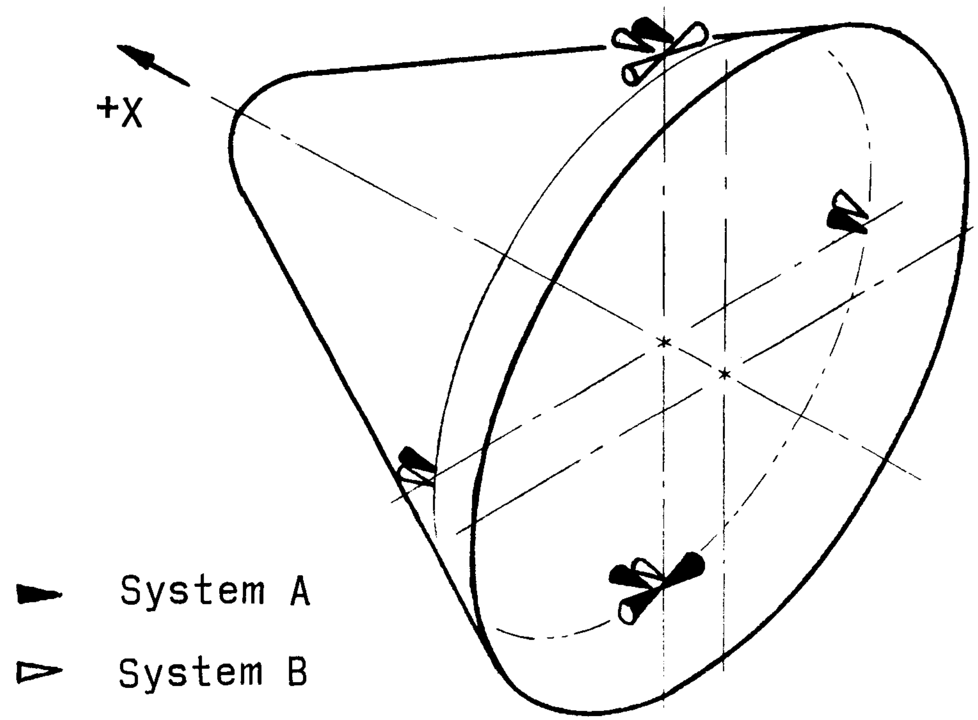


Figure 70.- Reaction control system - Command module - thrust chamber positions

[redacted]

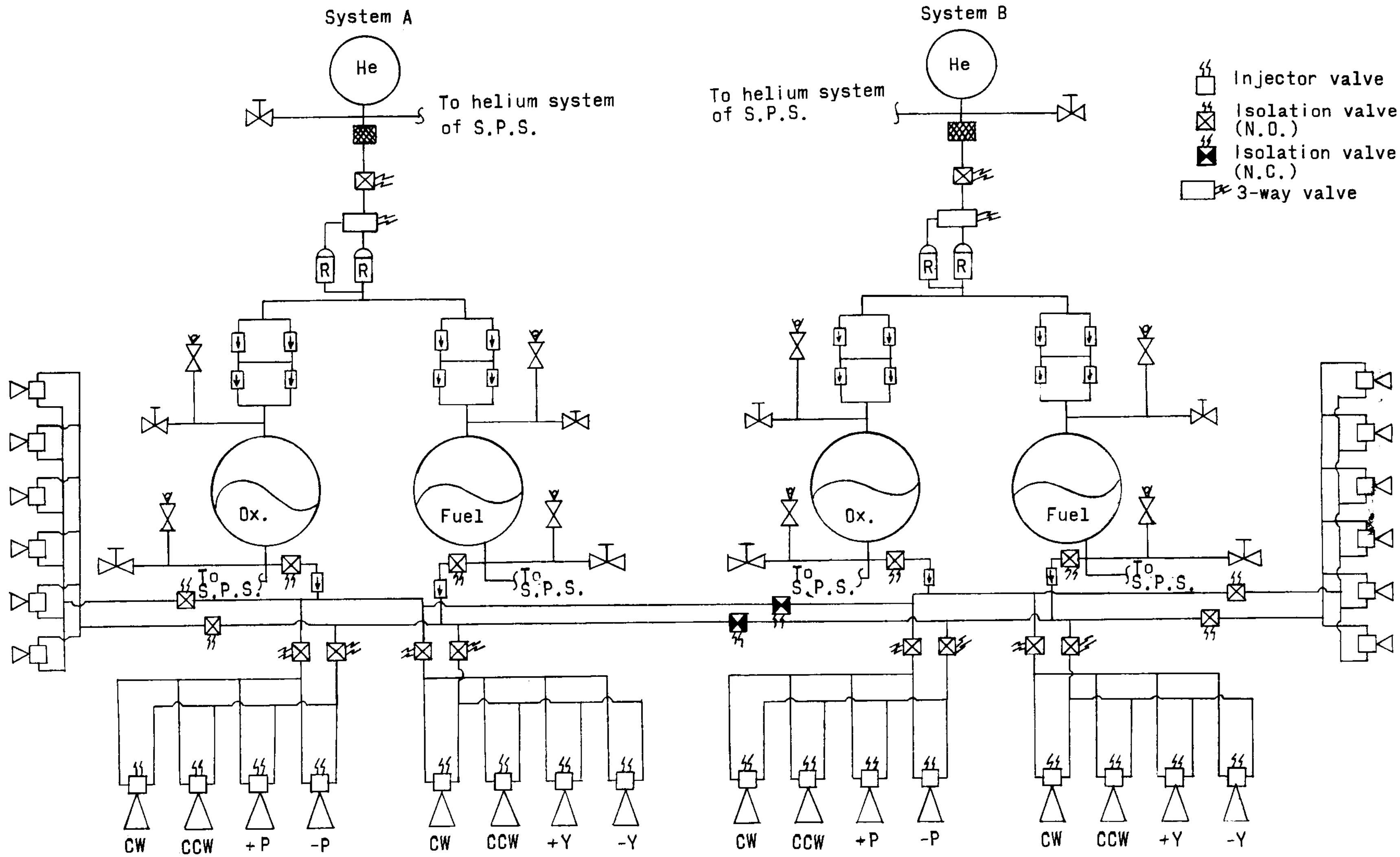


Figure 71.- Reaction control system, service module - schematic.

[redacted]

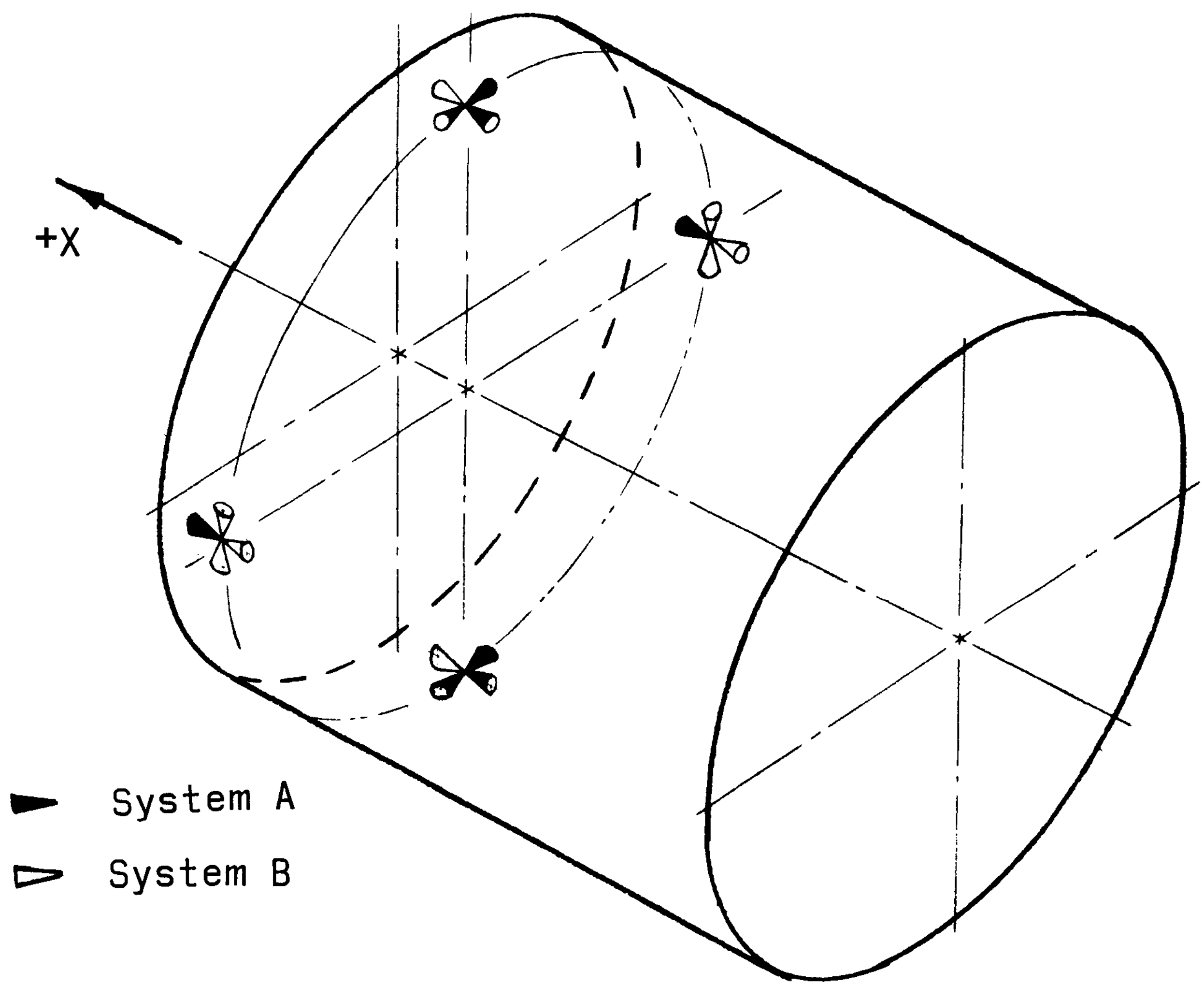
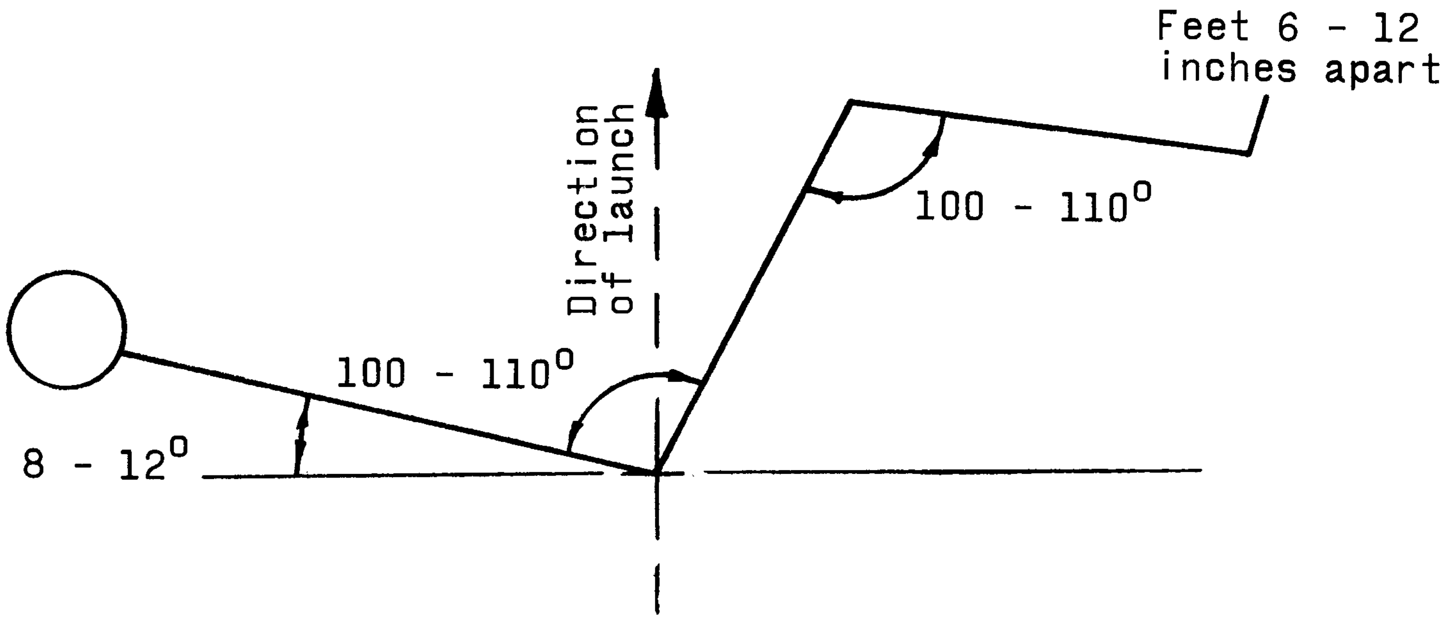


Figure 72.- Reaction control system, Service module, thrust chamber positions



Upper arm parallel to trunk

elbow interior angle approximately 135°

Figure 73.- Body angles

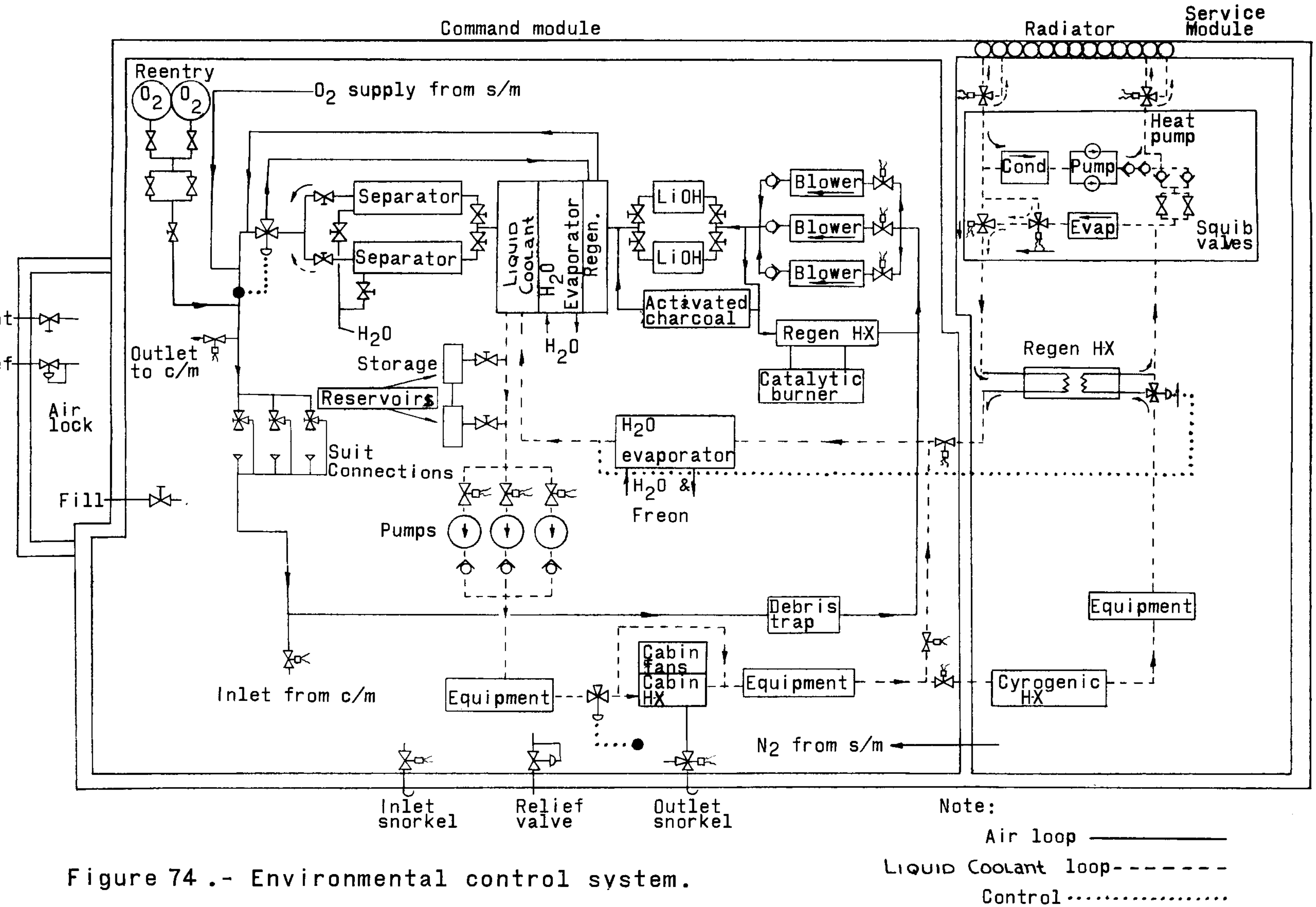


Figure 74 .- Environmental control system.

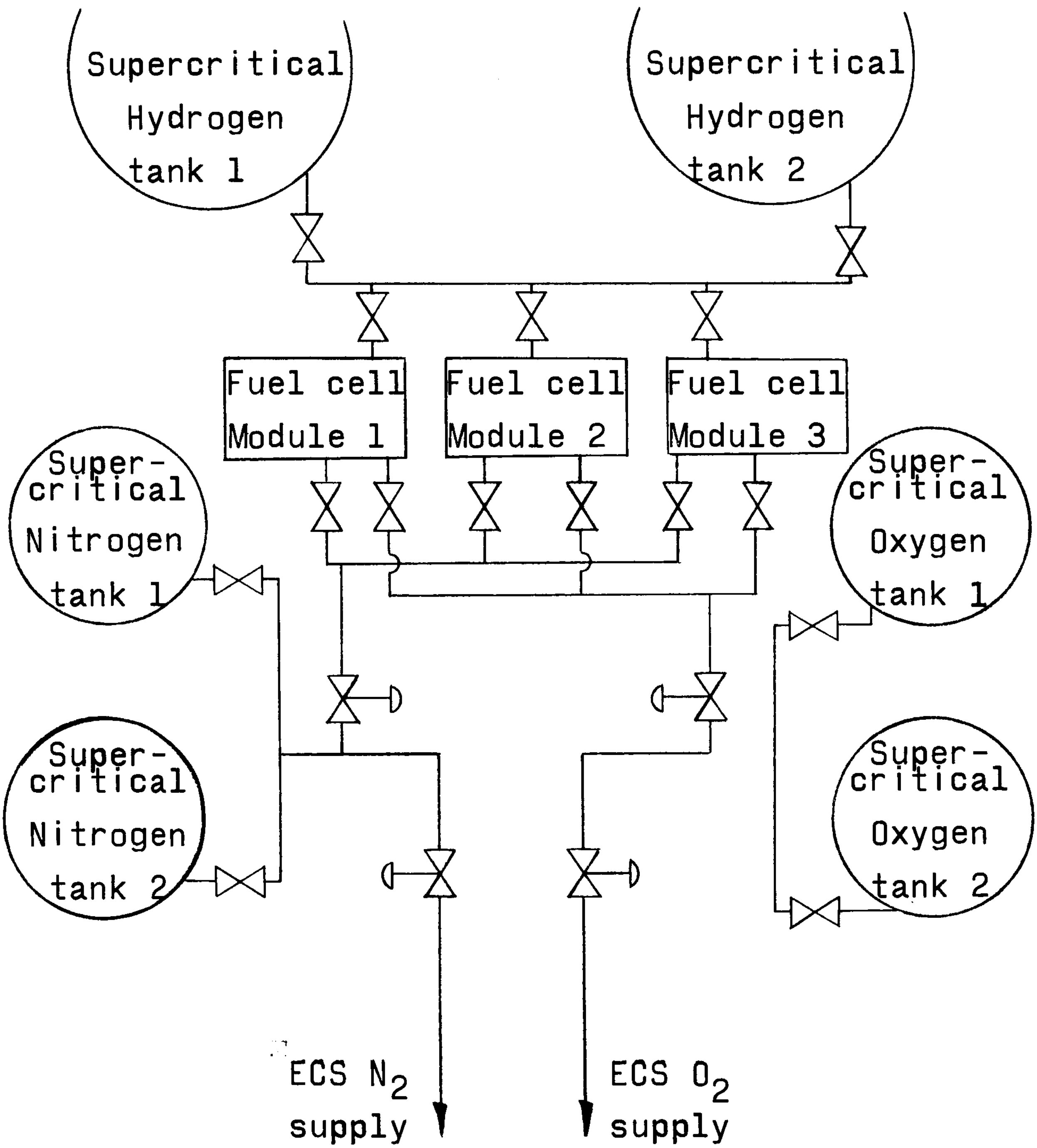


Figure 75.- Electrical power system - schematic.

[redacted]

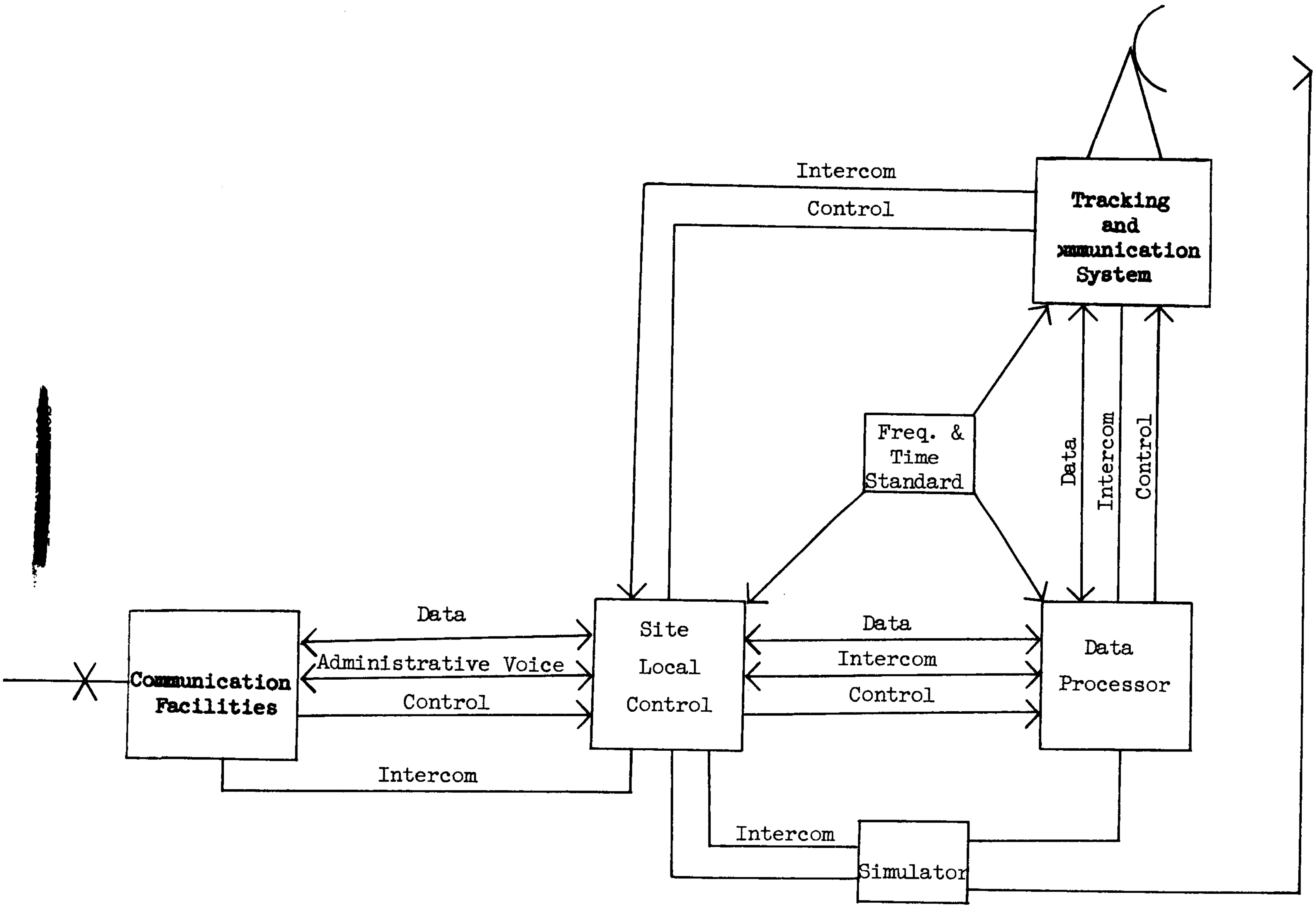


Figure 76.- Functional layout of GOSS station.

[redacted]

Data Lines
Tracking and Communications
Data Processor

Voice	←——→
TM (D)	——→
TM (U)	←——
Range	←——→
Range Rate	←——→
Azimuth	←——→
Elevation	←——→
Status	——→

Data Processor - Local Control

Voice	←——→
TM (D)	——→
TM (U)	←——
Range	←——→
Range Rate	←——→
Azimuth	←——→
Elevation	←——→
Status	——→

Local Control - Ground Communication Facility

Voice	←——→
TM (D)	——→
TM (U)	←——
Range	←——→
Range Rate	←——→
Azimuth	←——
Elevation	←——
Status	——→

LEGEND

←——→ = Two Way Communication
——→ = One Way Communication

Figure 77.- Data flow at GOSS station.

NASA-Langley, 1962